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OPERATING CHARACTERISTICS OF GAS-TURBINE AIRCRAFT ENGINES

by

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FOREIGN TECHNOLOGY DIVISION

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ABSTRACT

(U) The operating characteristics of aviation gas turbine engines are reviewed. The information covers thermodynamic control principles, throttling, altitude, speed, acceleration, startup, and special characteristics of turbojet, turboprop, and turbofan engines. A number of important problems are covered, which until recently were not included in textbooks for civil aviation institutes. These problems include specific characteristics of double shaft by-pass turbojet engines with a high by-pass ratio; the effect of losses along the ducts and external factors on the engine performance and parameters; compressor instability, noise, and engine reliability. The book is based on non-Soviet information on jet engines published in aviation periodicals, and also on the experience gained by the Soviet aviation industry and the Ministry of Civil Aviation with the operation of aviation gas turbine engines designed by A. G. Ivchenko; N. D. Kuznetsov, and P. A. Solov'yev. It is intended as a textbook for high scientific institutions of civil aviation, and it might be useful to persons engaged in aviation engine maintenance.

AFSC FORM AUG 66 4 (FTD OVERPRINT, DEC 66)

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U. S. BOARD ON GEOGRAPHIC NAMES TRANSLITERATION SYSTEM

Block	Italic	Transliteration	Block	Italic	Transliteration
А а	*А а*	A, a	Р р	*Р р*	R, r
Б б	*Б б*	B, b	С с	*С с*	S, s
В в	*В в*	V, v	Т т	*Т т*	T, t
Г г	*Г г*	G, g	У у	*У у*	U, u
Д д	*Д д*	D, d	Ф ф	*Ф ф*	F, f
Е е	*Е е*	Ye, ye; E, e*	Х х	*Х х*	Kh, kh
Ж ж	*Ж ж*	Zh, zh	Ц ц	*Ц ц*	Ts, ts
З з	*З з*	Z, z	Ч ч	*Ч ч*	Ch, ch
И и	*И и*	I, i	Ш ш	*Ш ш*	Sh, sh
Й й	*Й й*	Y, y	Щ щ	*Щ щ*	Shch, shch
К к	*К к*	K, k	Ъ ъ	*Ъ ъ*	"
Л л	*Л л*	L, l	Ы ы	*Ы ы*	Y, y
М м	*М м*	M, m	Ь ь	*Ь ь*	'
Н н	*Н н*	N, n	Э э	*Э э*	E, e
О о	*О о*	O, o	Ю ю	*Ю ю*	Yu, yu
П п	*П п*	P, p	Я я	*Я я*	Ya, ya

* ye initially, after vowels, and after ъ, ь; e elsewhere. When written as ё in Russian, transliterate as yё or ё. The use of diacritical marks is preferred, but such marks may be omitted when expediency dictates.

FOLLOWING ARE THE CORRESPONDING RUSSIAN AND ENGLISH DESIGNATIONS OF THE TRIGONOMETRIC FUNCTIONS

Russian	English
sin	sin
cos	cos
tg	tan
ctg	cot
sec	sec
cosec	csc
sh	sinh
ch	cosh
th	tanh
cth	coth
sch	sech
csch	csch
arc sin	$\sin^{-1}$
arc cos	$\cos^{-1}$
arc tg	$\tan^{-1}$
arc ctg	$\cot^{-1}$
arc sec	$\sec^{-1}$
arc cosec	$\csc^{-1}$
arc sh	$\sinh^{-1}$
arc ch	$\cosh^{-1}$
arc th	$\tanh^{-1}$
arc cth	$\coth^{-1}$
arc sch	sech^{-1}
arc csch	csch^{-1}
rot	curl
lg	log

The book is dedicated to the discussion of operating characteristics of gas-turbine aircraft engines — an important section of the general course of aircraft engine theory. In it are discussed thermodynamic fundamentals of control, throttle, altitude, high-speed, acceleration, starting, and also special characteristics of turbojet, turboprop and turbofan engines.

In the book emphasis is placed on consideration of a number of important questions, which until recently were little covered in educational literature for aviation technical institutes: peculiarities of characteristics of dual-shaft ducted-fan engines, including at high bypass ratio; appraisal of the effect of losses through the air-gas passage, and also external conditions (including the humidity of the atmosphere) on engine operation and parameters; methods of combating unstable compression operation; noise level characteristics; concise information on the operational reliability of the engine.

Characteristics of gas-turbine aircraft engines are discussed basically in reference to subsonic passenger aircraft, equipped with engines with unregulated intakes and jet nozzles. Therefore, characteristics of these elements of jet engines are not examined in the book.

While writing the book the author widely used technical information materials on foreign aircraft engines, published in aviation in periodicals, and also accumulated experience in the aviation industry and in flight subdivisions of the Ministry of Civil Aviation on the operation of domestic gas-turbine aircraft engines of the design of A. G. Ivchenko, N. D. Kuznetsov and P. A. Solov'yev.

The book assumes that readers have knowledge of thermodynamics, turbomachine theory and general theory of jet engines.

The book is intended as a training aid for higher educational institutions of civil aviation. It can be useful for persons working in the field of operation and repair of aircraft engines. The book has 151 figures, 4 tables, 23 bibliographies.

Critic: Prof.
I. I. Kulagin, honored
scientist and technician
of RSFSR, Doctor of
technical sciences.

BASIC ABBREVIATIONS, DESIGNATIONS, SUPERSCRIPTS AND SUBSCRIPTS

Abbreviations

[TRD] (ТРД) – turbojet engine

[TRDF] (ТРДФ) – turbojet engine with afterburner

[DTRD] (ДТРД) – ducted-fan engine

[DTRDF^{II}] (ДТРДФ^{II}) – ducted-fan engine with afterburner in the secondary duct

[DTRDF^{I+II}] (ДТРДФ$^{I+II}$) – ducted-fan engine with afterburners in both ducts

[DTRD(P)] (ДТРД(П)) – ducted-fan engine with forward location of fan

[DTRD(Z)] (ДТРД(З)) – ducted-fan engine with aft location of fan

[TVlD] (ТВлД) – turbofan engine

[TVD] (ТВД) – turboprop engine

[GTD] (ГТД) – gas-turbine engine

[LRR] (ЛРР) – line of operating conditions

[TVA] (ТВА) – turbofan unit

Basic Designations

p, T, γ, ρ – pressure, temperature, specific gravity, density of gas, respectively

V, c_0 – flight speed

H – flight altitude

R – reaction thrust, gas constant

$R_{уд}$ – specific thrust

$C_{уд}$ – specific fuel consumption

$R_{лоб}$ – frontal thrust

$\gamma_{ДВ}$ — weight per horsepower

π — compression (expansion) ratio

Δ — degree of gas preheating per cycle

G_B — air mass flow rate

$G_Г$ — gas mass flow rate

G_T — fuel mass flow rate

x — degree of energy exchange

y — bypass ratio (coefficient of distribution of air between ducts)

$N_К$, N_T — shaft horsepower of compressor and turbine respectively

n — number of revolutions; polytropic exponent

u — peripheral velocity

m_T — relative fuel consumption

$\xi_{К.С}$ — combustion efficiency

$L_{ад}$ — adiabatic work

$L_{пол}$ — polytropic work

L_r — work of friction

L_T, $L_К$, L_p, L_c — work of turbine, compressor, expansion, compression, respectively

L_e — effective work of cycle

$q_{ВН}$ — thermochemical energy of fuel, pertaining to 1 kg of air

q_I — heat imparted to 1 kg of air (supplied heat)

q_{II} — removed heat

η — efficiency

$\eta_К^*$, η_T^*, η_p, η_c — efficiency of compressor, turbine, expansion, compression, respectively

η_B — propeller efficiency

η_e — effective efficiency

η_R — thrust efficiency

η_O — overall efficiency

c — speed of gas

a — speed of sound

D — diameter

f — area of cross section

c_p, c_v — specific heat of gas at constant pressure and at constant volume, respectively

$k = c_p/c_v$ — adiabatic index

φ – speed ratio, angle of rotation of stator
i – enthalpy
β – air (gas) bleed factor
σ* – total pressure recovery factor
Π(λ), q(λ), μ(λ) – gas-dynamic functions

$$e = \pi^{\frac{k-1}{k}}$$

$$\varepsilon = 1 - \frac{1}{\pi^{\frac{k-1}{k}}}$$

Superscripts

* – parameters of retarded flow.
I, II – parameters of primary and secondary ducts.

Subscripts

ф – boost
к, т – compressor, turbine
пр – given
р, с – expansion, compression
в – propeller
R – thrust
e – effective
р.с – jet nozzle
к.с – combustion chamber
ф.к – afterburner
НА – guide vanes
СА – nozzle box
см – mixing
к.см – mixing chamber
расч – calculated
ид – ideal
лоб – frontal
НД – low pressure

ВД — high pressure
кр — critical
э — effective, ejector
д — diffuser, dynamic
0 — available, initial, during ground operation
Н — external
в — internal
ад — adiabatic
пол — polytropic
уд — specific
дв — engine
в.с — exhaust system

Basic Flow Areas

Н, 0 — area of undisturbed flow
1 — compressor inlet
2 — compressor outlet
3 — combustion chamber outlet
4 — turbine exhaust
$4_ф$ — afterburner outlet
5, $5_ф$ — jet nozzle outlet

CHAPTER I

GENERAL INFORMATION ON GAS-TURBINE AVIATION ENGINES AND THEIR CHARACTERISTICS

§ 1. Classification of Gas-Turbine Engines

An engine consisting of air compressor, combustion chamber and gas turbine is called a *gas-turbine engine* (GTD). The gas turbine is for rotating the compressor, with which the pressure of the working medium (air), entering the combustion chamber, is increased; the gas turbine also transmits excess mechanical energy to the user. Thus, the gas turbine is the controlling element of the GTD.

The gas-turbine engine found the widest application in aviation for the creation of propulsion force of aircraft – thrust. Thrust of gas-turbine aircraft engines appears with outflow of gases from the engine nozzle – by means of so-called *direct reaction*. It can also be obtained by means of *indirect reaction* with the transmission of mechanical energy from the gas turbine, for example, to the propeller, which while rotating rejects back large masses of air; in this case there appears oppositely directed propulsion force – propeller thrust.

Gas-turbine engine, for which thrust is formed as a result of direct reaction, is called a turbojet engine (TRD). Power of the TRD turbine is always equal to the power of the compressor:

$$N_{т}=N_{к}. \tag{1.1}$$

For a TRD the energy of excess gas pressure behind the turbine is expended on expansion and outflow of the gas at high speed.

For a gas-turbine engine in which thrust is obtained by indirect reaction, the power of the gas turbine exceeds that of the compressor and its excess portion is transferred to the user. Such energy users of the turbine can be the propeller, fan or compressor of the secondary duct:

$$\left.\begin{aligned} N_{\text{т}}&=N_{\text{к}}+N_{\text{в}};\\ N_{\text{т}}&=N'_{\text{к}}+N_{\text{вл}};\\ N_{\text{т}}&=N_{\text{кI}}+N_{\text{кII}}. \end{aligned}\right\} \qquad (1.2)$$

In accordance with this the GTD is called <u>turboprop</u> (TVD), <u>turbofan</u> (TV1D) and <u>ducted-fan engine</u> (DTRD).

Gas-turbine aircraft engines of indirect reaction have two propelling agents: as the first we have the basic engine duct, consisting of air intake, compressor, combustion chamber, turbine and jet nozzle; as the second we have either a propeller (TVD), or a secondary engine duct, including air intake, compressor or fan and jet nozzle (TV1D or DTRD), and sometimes an afterburner.

Classification of gas-turbine aircraft engines, principle of action of which was examined above, is shown in Fig. 1.1.

Turbojet engines (single-shaft and two-shaft) can be equipped with afterburners, which permit substantially increasing takeoff and especially flight thrust; the greater the flight thrust, the higher the flight speed is.

For thrust augmentation of turboprop engines we use injection of water into the compressor. Water injection can be successfully applied for TRD (DTRD), however, it is less effective than additional burning of fuel after the turbine.

The gas-turbine engines shown in Fig. 1.1 can be considered as particular cases of ducted-fan engines. These engines are

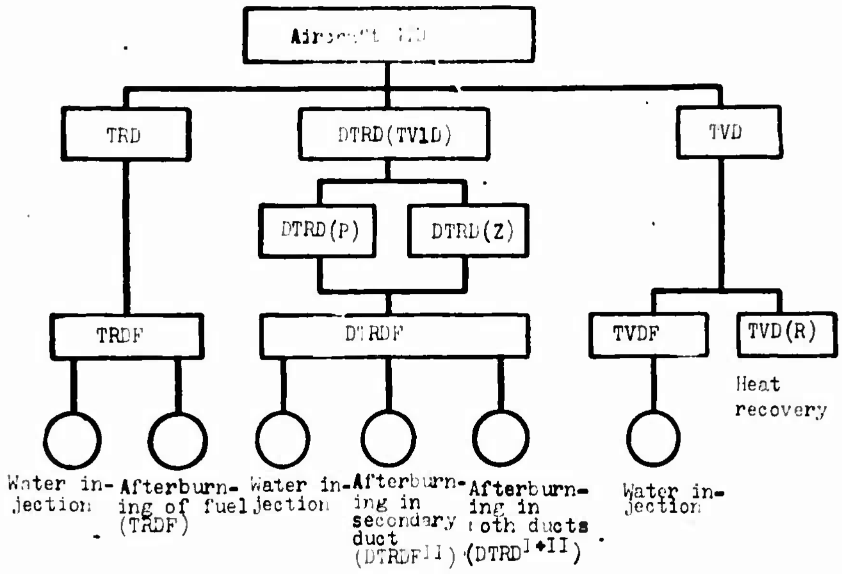

Fig. 1.1. Classification of gas-turbine aircraft engines.

distinguished by the ratio of the amount of air passing through the secondary and primary ducts, so-called bypass ratio:

$$y=\frac{G_{aII}}{G_{aI}},$$

and also the structural scheme and principle of formation of thrust in the secondary duct. If thrust of an aircraft or helicopter propeller is created as a result of the direct imparting of acceleration to the rejected masses of air, then thrust of the secondary duct of a ducted-fan engine will be formed as a result of accomplishment of a thermodynamic cycle, the most important process of which is increase of pressure of the working medium in the compressor or fan. Work of this cycle is finally used for acceleration of flow and, consequently, the creation of thrust.

The bypass ratio for different types of GTD has the following values: TRD y = 0; TDRD y = 0.25-1.5; TV1D y = 2-10; aircraft TVD y = 50-100; helicopter TVD y = 200-700.

On Figs. 1.2, 1.3 and 1.4 there are shown typical diagrams of contemporary gas-turbine aircraft engines.

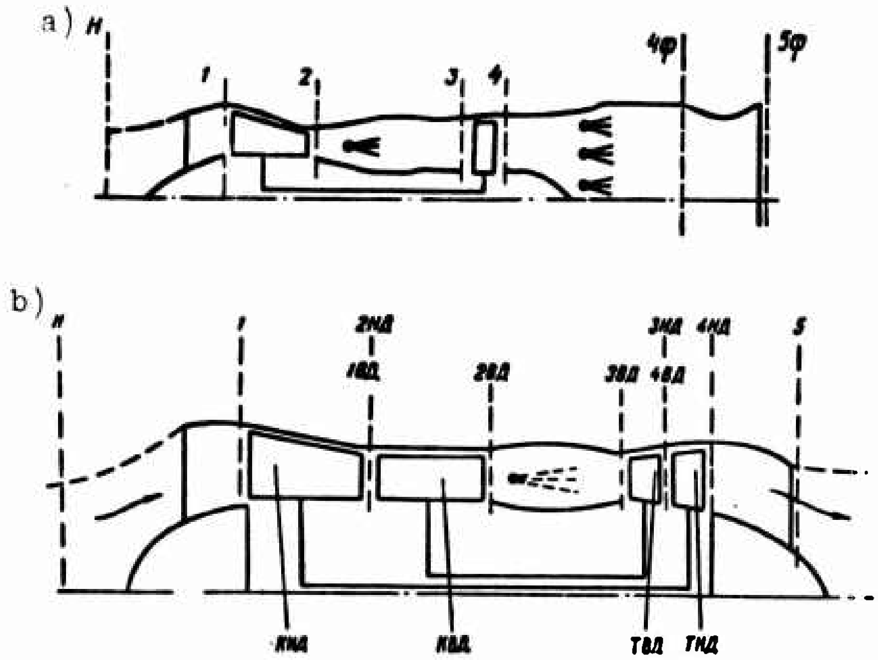

Fig. 1.2. Diagram of ducted-fan engines: a) single-shaft; b) two-shaft. [Translator's Note: НД = low pressure; ВД = high pressure; КНД = low pressure compressor; КВД = high pressure compressor; ТВД = high pressure turbine; ТНД = low pressure turbine].

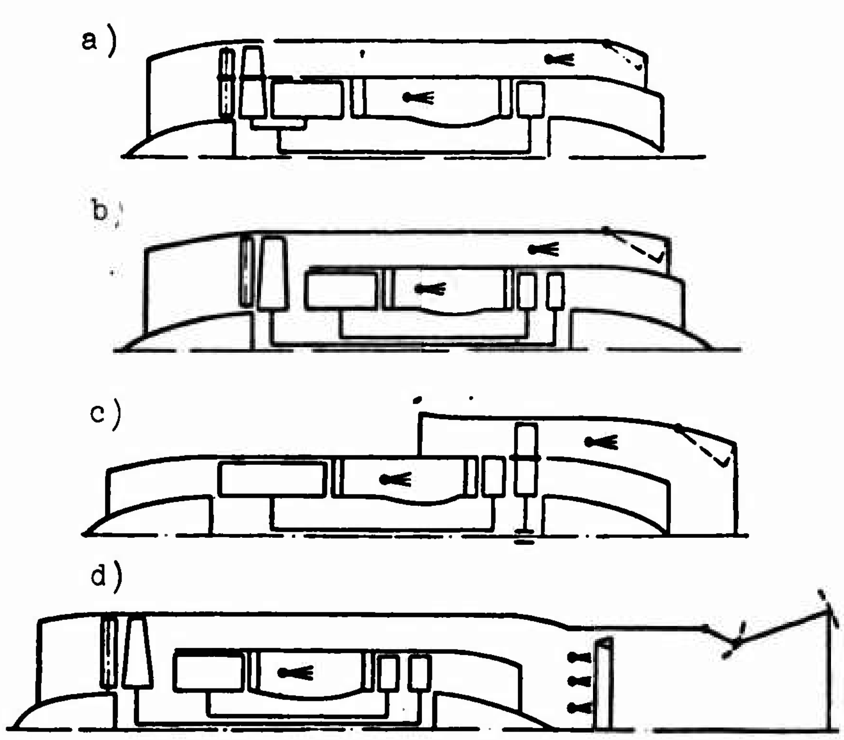

Fig. 1.3. Diagram of ducted-fan engines: a) single-shaft with separate exhaust; b) two-shaft with separate exhaust; c) two-shaft with aft location of fan; d) two-shaft with mixing chamber and afterburner.

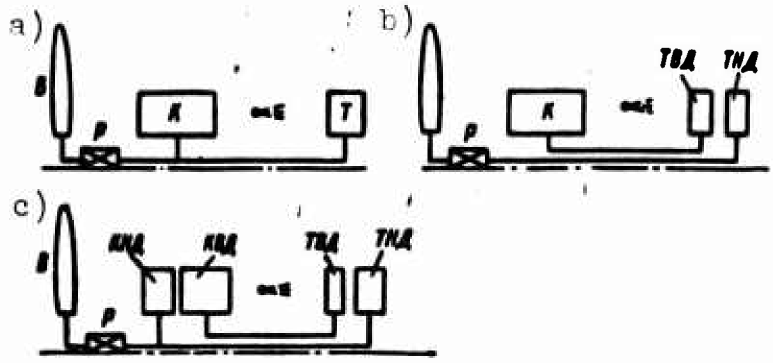

Fig. 1.4. Diagrams of turboprop engines: a) single-shaft; b) two-shaft with single-stage compressor; c) two-shaft with split compressor.

§ 2. The Concept of Operating Characteristics of Aircraft Engines

Relationships showing the effect of operating conditions on basic parameters are called the operating characteristics of gas-turbine aircraft engines: thrust, power, specific and hourly consumption of fuel.

Operating conditions include: state of the external atmosphere (pressure, temperature, humidity of air), flight parameters (altitude and speed of aircraft), program of engine control, determining the selected regularity of change of controllable parameters depending on the position of control elements, operating peculiarities of the engine on the aircraft.

Operating conditions determine the performance of the engine. The latter is evaluated by performance parameters, which include both internal parameters (factors) — number of engine revolutions, position of the regulator of the jet nozzle throat area (exit section) and others, and external — flight conditions, condition of external atmosphere.

Engine efficiency is determined not only by its basic parameters, but also by such important operational properties as the tendency toward unstable operation, level of noise produced by the engine, reliability probability, engine life.

Operating conditions essentially affect the reliability of the engine, causing mechanical disturbances and deformation of the gas-air duct of the engine and its elements (warping, denting, corrosion, carbon formation, icing), maladjustment of separate subassemblies. This leads to drop of efficiency of the subassemblies, causes deterioration of basic engine indices, and disturbs its thermal balance.

Forms of Operating Characteristics

Operating characteristics of gas-turbine aircraft engines are subdivided into throttle, high-speed, altitude, and also special.

Throttle characteristics represent the relationship of basic engine parameters (thrust, specific fuel consumption, etc.) to its performance parameters (for example, number of revolutions of engine rotor).

High-speed and altitude characteristics of the engine show change of its basic parameters depending on flight conditions (speed and altitude) at a prescribed program of engine control.

Sometimes altitude and high-speed engine characteristics are represented in accordance with a concrete flight profile of the aircraft. Such characteristics are very graphic, since they show how the necessary thrusts (powers) of aircraft and available thrusts (powers) of engines are coordinated in flight.

Throttle, altitude and high-speed engine characteristics must consider possible physical limitations, which appear in engine operation.

Limitations are stipulated by:

strength of components (subassemblies);
the appearance of maximum permissible gas temperatures;
the appearance of unstable operating conditions;

the appearance of conditions of gas-dynamic "blocking" of the gas-air duct;

the appearance of conditions of natural oscillations and vibrations.

Special engine characteristics include characteristics of the noise level, stability margin, operational reliability, technical and economic characteristics.

Methods of Obtaining Engine Characteristics

Engine characteristics are obtained experimentally — by bench tests of the engine or on so-called flying laboratories in flight. They can also be obtained by calculation (analytically).

Usual testing stands, which aircraft production or repair plants are equipped with permit obtaining throttle (stand) characteristics of the engine. Then these characteristics are reduced to standard atmospheric conditions ($p = 760$ mm Hg and $T = 288^{\circ}K$ when $M_0 = 0$ and $H = 0$).

For obtaining altitude and high-speed characteristics it is necessary to have special stands with pressure chambers, which permit simulating prescribed flight conditions, i.e., reproducing the necessary values of pressure and temperature of gas at the engine inlet and exit. Such stands are very complex, bulky and expensive devices. Expenditures of huge energy power are required for actuating them, therefore, they are installed only in scientific, research and constructional organizations.

The most accessible method of obtaining altitude and high-speed engine characteristics is the use of a flying laboratory. An example of such a laboratory is the English aircraft "Ashton," created on the basis of the "Vulcan" bomber and equipped with four "Avon" turbojet engines. As a flying laboratory the United States intends to use the supersonic experimental bomber B-70, designed for flights at a speed corresponding to $M = 3$.

Deficiencies of the flying laboratories include their complexity and high costs, and also the limitedness of range of altitude and high-speed tests, which is determined by mechanical and flying characteristics of the aircraft-laboratory. It is obvious that the new (tested) engine cannot be tested at flight speeds and altitudes that are not accessible to the aircraft-laboratory.

The calculation (analytic) method of obtaining engine characteristics at present has reached a high degree of perfection, especially in connection with the application of high-speed electronic computers. Engine characteristics, obtained in such a way, give a small degree of error, as a rule not exceeding 2-3%.

Program of Engine Control

The program of control of a gas-turbine engine is the regularity of change of engine parameters, determined by the position of control elements. This regularity ensures the most advantageous engine characteristics.

Examples of control programs are programs of control for maximum thrust (power) of the engine, its best economy, preservation of complete similarity of turbocompressor operation, minimum noise level of the engine at assigned thrust, etc.

Controlling Elements and Controllable Parameters

Operating conditions of GTD are established with the aid of control elements. They include first of all the fuel feed control, interlinked usually with the speed regulator. By affecting the fuel feed control with the engine control lever (RUD), it is possible to change the flow rate of fuel per second fed to the combustion chamber, and consequently, the number of revolutions of the rotor; in the case of a TRD with fixed geometry it is possible to change the temperature of gas before the rotor turbine. Thus, the control element (fuel feed control) affects the performance parameter — number of revolutions of rotor — through the regulating factor (flow rate of fuel per second).

At fixed position of the RUD the speed regulator maintains constant revolutions of the engine rotor. If the engine has two independent control elements (for example, fuel feed control and jet nozzle control), and consequently, two regulating factors (flow rate of fuel per second and jet nozzle throat or exit section), then the number of independently controllable performance parameter is also equal to two: the number of revolutions of the engine rotor and the gas temperature before the turbine. The number of controllable parameters is always equal to the number of regulating factors.

Besides the speed regulator, contemporary gas-turbine engines are equipped with a number of other automatic machines, which control these or other performance parameters. In this case, the gas-turbine aircraft engine is more complicated, the more automatic devices on it. On the engine are installed automatic machines that regulate the position of the jet nozzle, the guide vanes of the compressor, change of gas temperature before the turbine, propeller pitch, etc.

CHAPTER II

THROTTLE CHARACTERISTICS OF TRD

The relationship of thrust and specific fuel consumption on the number of engine revolutions at a predetermined program of control is called throttle characteristics of TRD. These characteristics usually supplement the curve of fuel consumption per hour, and also the curve of gas temperature change past the turbine. The latter gives the possibility of judging the reliability probability of operation of the combustion chamber, turbine and jet nozzle of the engine. Thus, throttle characteristics are depicted in the form of curves:

$$R=f(n);\ C_{yд}=f(n);\ G_т=f(n);\ T_4^*=f(n).$$

In certain cases the throttle characteristics in the form of relationships of specific fuel consumption and relative thrust:

$$C_{yд}=f(\bar{R}),$$

where

$$\bar{R}=\frac{R}{R_{макс}}.$$

The number of engine revolutions are changed by changing the fuel feed to combustion chamber. The latter is carried out by movement of the RUD.

With increase of fuel feed to the combustion chamber the gas temperature at the turbine inlet increases, during which the power of the turbine is increased and becomes greater than compressor power:

$$N_т > N_к.$$

Excess turbine power, equal to $\Delta N = N_т - N_к$, is expended for acceleration of rotation of the engine turbocompressor, i.e., for increase of its revolutions. Increase of revolutions is ceased when under certain conditions there is again established equality of powers:

$$N_т = N_к.$$

If we decrease the fuel feed to the combustion chamber, the process will be reversible. Now the changed turbine power will even be less than the compressor power:

$$N_т < N_к.$$

The unbalance of powers is eliminated by lowering the engine revolutions to their new equilibrium value.

Programs of control during throttling provide for regularities of change of cross sections of the gas-air duct of the engine depending on the number of revolutions (for example, adjustment of jet nozzle or guide vanes). These regularities will be selected so as to assure good economy of engine operation at lowered performance, to improve the operating properties of the engine (for example, to increase the surge margin, to improve the accelerating capacity of TRD). In a particular case the control program of TRD provides fixed geometry of the engine, i.e., observance of conditions: $f_5 = \text{const}$, $\varphi_{НА} = \text{const}$, etc.

§ 1. Combined Work of the Compressor and Turbine in TRD

Basic Equations

Characteristics of TRD are determined to a considerable extent by the combined work of compressor and turbine.

Below are examined fundamental equations of the combined work of compressor and turbine of TRD:

1) flow rate equation, or equation of material balance. This equation establishes the relationship between flow rates per second of air at the compressor inlet and gas through the turbine:

$$G_{\text{г}}=G_{\text{в}}-G_{\text{от}}+G_{\text{т}}, \qquad (2.1)$$

where $G_{\text{в}}$ — the quantity of air entering the compressor per second; $G_{\text{г}}$ — the quantity of gas passing through the turbine per second; $G_{\text{от}}$ — the quantity of air withdrawn from the compressor per second for cooling the turbine, driving the aircraft units, etc.; $G_{\text{т}}$ — the quantity of fuel fed to the combustion chamber.

Expression (2.1) can be written:

$$G_{\text{г}}=G_{\text{в}}\left(1+\frac{G_{\text{т}}-G_{\text{от}}}{G_{\text{в}}}\right)=\beta G_{\text{в}}, \qquad (2.2)$$

where $\beta=\frac{G_{\text{г}}}{G_{\text{в}}}\gtrless 1$.

2) Equation of power balance (equation of energy balance). It has the following form:

$$N_{\text{т}}=\frac{N_{\text{к}}}{\eta_m}, \qquad (2.3)$$

where η_m — mechanical efficiency, considering friction in turbocompressor bearings, and also expenditures of power for driving the units (fuel pump, tachometer, generator, oil pumps); usually $\eta_m=0{,}99\div0{,}995$.

By replacing power with work in equation (2.3), we obtain <u>equation of work balance</u>

$$L_{\text{т}} G_{\text{г}} = \frac{L_{\text{к}} G_{\text{в}}}{\eta_m}$$

or

$$L_{\text{т}} = \frac{L_{\text{к}}}{\beta \eta_m}. \tag{2.4}$$

3) <u>Equation of equality of revolutions</u>. For turbocompressors of contemporary TRD

$$n_{\text{т}} = n_{\text{к}} = n. \tag{2.5}$$

4) The relationship between the <u>compression ratio of compressor</u> and expansion ratio of turbine:

$$\pi^*_{\text{т}} = \frac{\pi_{\text{д}} \pi^*_{\text{к}} \sigma^*_{\text{к.с}}}{\pi_{\text{р.с}}}, \tag{2.6}$$

where

$$\pi_{\text{д}} = \frac{p^*_1}{p_H}; \quad \pi^*_{\text{к}} = \frac{p^*_2}{p^*_1};$$

$$\sigma^*_{\text{к.с}} = \frac{p^*_3}{p^*_2}; \quad \pi_{\text{р.с}} = \frac{p^*_4}{p_H}.$$

Equation of Line of Operating Conditions

By using equations (2.2) and (2.4), it is possible to formulate a generalized equation of the line of operating conditions of TRD in the form of relationship

$$\pi^*_{\text{к}} = f[q(\lambda)]$$

and to apply it on the characteristic of TRD compressor.

Let us write the relationship between the flow rates of air for the compressor inlet section and of gas for the nozzle throat of the turbine:

$$\beta G_1 = G_{CA}$$

or

$$m_a \frac{p_1^*}{\sqrt{T_1^*}} f_1 q(\lambda_1)\beta = m_r \frac{p_{CA}^*}{\sqrt{T_{CA}^*}} f_{CA} q(\lambda_{CA}). \qquad (2.7)$$

Considering that

$$T_1^* = T_H^*; \quad T_{CA}^* = T_3^*,$$
$$p_{CA}^* = p_2^* \sigma_{CA}^* \sigma_{K.C}^*,$$

let us convert equation (2.7). We obtain

$$\pi_K^* = \frac{p_2^*}{p_1^*} A q(\lambda_1) \sqrt{\frac{T_3^*}{T_H^*}} \beta, \qquad (2.8)$$

where

$$A = \frac{f_1 \frac{m_a}{m_r}}{f_{CA} q(\lambda_{CA}) \sigma_{CA}^* \sigma_{K.C}^*} = $$
$$= \text{const}. \qquad (2.9)$$

If we accept $\beta = 1$; $\frac{m_a}{m_r} = 1$; $q(\lambda_{CA}) = 1$; $\sigma_{CA} = 1$ and $\sigma_{K.C}^* = 1$,

then

$$A \approx \frac{f_1}{f_{CA}}.$$

Equation (2.8) represents the flow rate equation. On the compressor characteristic in coordinates π_K^* and $q(\lambda_1)$ this equation is depicted by a family of straight lines (Fig. 2.1) for different values of $\Delta = \sqrt{\frac{T_3^*}{T_H^*}}$, constant for the given engine. The greater Δ is (i.e., the greater T_3^* and the smaller T_H^*), the greater the inclination of the line.

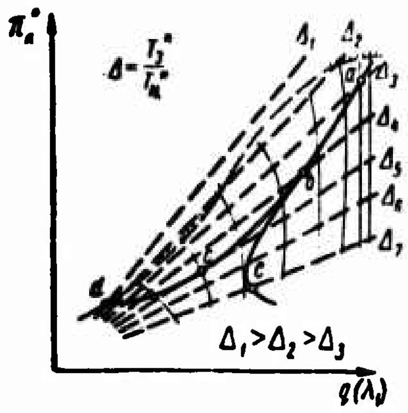

Fig. 2.1. Family of lines Δ = const on the compressor characteristic.

Let us now convert the equation of work balance (2.4). We have

$$\beta\eta_{m}118\,T_3^*\varepsilon_T^*\eta_{T}^*=102{,}5\,T_H^*(e_K^*-1)\frac{1}{\eta_K^*},$$

where $e_K^*=\pi_K^{*0,286}$; $\varepsilon_T^*=1-\frac{1}{\pi_T^{*0,25}}$.

From this expression we find

$$\frac{T_3^*}{T_H^*}=\frac{102{,}5}{118}\cdot\frac{(e_K^*-1)}{\beta\varepsilon_T^*\eta_{T}^*\eta_{K}^*\eta_{m}}. \tag{2.10}$$

Let us place the value of $\frac{T_3^*}{T_H^*}$ in equation (2.8).

After simple conversions we obtain

$$\frac{\pi_K^*}{\sqrt{\frac{\pi_K^{*0,286}-1}{\eta_K^*}}}=Aq(\lambda_1)\sqrt{\frac{102{,}5}{118}\cdot\frac{\beta}{\varepsilon_T^*\eta_T^*}}. \tag{2.11}$$

Expression (2.11) is called the equation of line of operating conditions.

Let us assume that $\beta=1$, $\eta_K^*=\text{const}$, $\eta_T^*=\text{const}$ and $\pi_T^*=\text{const}$ (i.e., λ_5 = 1). Then the equation of line of operation conditions will be depicted on the compressor characteristic in the form of parabolic curve a-b-e

with inflection point at $\pi_{\kappa}^{*}=1,75$.[1]

Along the given curve the gas temperature before the turbine T_3^* continuously drops. This is in complete conformity with equation (2.10) under the conditions stipulated above ($\beta=1$; $\pi_{T}^{*}=\text{const}$; $\eta_{T}^{*}=\text{const}$; $\eta_{\kappa}^{*}=\text{const}$).

If, however, we consider the actual regularity of change of π_{T}^{*}, and also the drop of η_{T}^{*} and η_{κ}^{*} in the region of low revolutions, then the real line of operating conditions will be depicted by curve a-b-c-d. Along this line the gas temperature T_3^* at first drops, reaches minimum at point c, and then continuously rises.

Line of operating conditions a-b-c-d is conditionally designated f_5 = const, since it is constructed on the assumption that the jet nozzle exit section is not regulated.

---

[1]We have

$$\frac{\pi_{\kappa}^{*}}{\sqrt{\pi_{\kappa}^{*0,286}-1}}=\beta q(\lambda_1).$$

We find π_{κ}^{*} from condition $\frac{dq(\lambda_1)}{d\pi_{\kappa}^{*}}=0$. In this case

$$\sqrt{\pi_{\kappa}^{*0,286}-1}-\frac{\pi_{\kappa}^{*}\cdot 0,286}{2\sqrt{\pi_{\kappa}^{*0,286}-1}\cdot\pi_{\kappa}^{*\frac{1}{k}}}=0.$$

By converting the given expression, we obtain

$$2\pi_{\kappa}^{*0,286}-2=0,286\pi_{\kappa}^{*0,286}.$$

Whence

$$\pi_{\kappa}^{*}=\left(\frac{2}{2-\frac{k-1}{k}}\right)^{\frac{k}{k-1}}=\left(\frac{2}{1,714}\right)^{3,5}\approx 1,75.$$

The Effect of Control of the Turbine Jet Nozzle and Nozzle Box on the Turbocompressor Operating Conditions at $n_{np} = \text{const}$

With increase of throat (exit section) of the jet nozzle the performance point of turbocompressor is shifted along the pressure characteristic of the compressor ($n_{np} = \text{const}$) to the region of lowered values of π_K^* and T_3^*. With decrease of the section this point is displaced to the region of raised values of π_K^* and T_3^*.

Really, with increase of the jet nozzle exit section the back-pressure behind the turbine is lowered, and pressure drop on the turbine increases.[1] Work of the turbine at the prescribed number of revolutions becomes greater than the work of compressor. This should have led to increase of revolutions, but since the speed regulator maintains n = const, it reduces fuel feed to the combustion chamber, as a result of which the gas temperature before the turbine T_3^* drops. Lowering of T_3^* leads to increase of the air flow rate through the compressor and to lowering of the compression ratio. Due to this, the performance point of the compressor will shift along line $n_{np} = \text{const}$ of the compressor from point a to point b (Fig. 2.2).

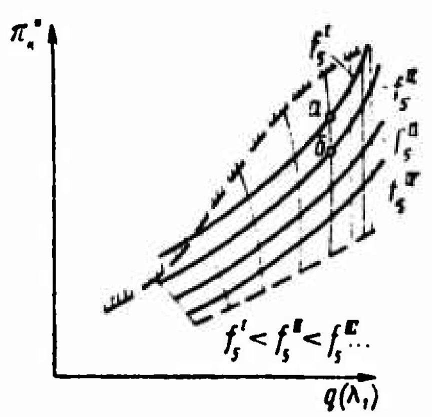

Fig. 2.2. Family of lines of operating conditions when f_5 = const.

The single line of operating conditions corresponds to each value of f_5 = const. With increase of the jet nozzle area the line

[1]See equation (2.13).

of operating conditions is shifted equidistantly to the region of reduced values of π_{k}^{*} and T_3^*. With reduction of the jet nozzle area it is shifted to the region of raised values of π_{k}^{*} and T_3^*.

Adjustment of the nozzle box throat is carried out by synchronous rotation of its blades by a special mechanism (Fig. 2.3). The

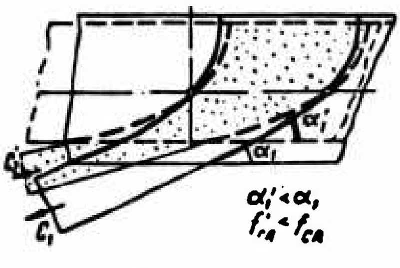

Fig. 2.3. Diagram of the variable-area nozzle box of the turbine.

connection between the nozzle box exit angle α_1 and the nozzle box throat is expressed by the following equation

$$f_{kp} = th \sin \alpha_1,$$

where t — blade pitch of nozzle box (CA); h — blade height of nozzle box.

With decrease of α_1 the value of f_{kp} is decreased and conversely.

From the flow rate equation (2.7) it follows that increase of f_{CA} decreases the pressure drop on the turbine π_{T}^{*}. Actually, the greater angle α_1 is, the smaller the peripheral component of exit velocity from the nozzle box of the turbine $c_{1u} = c_1 \cos \alpha_1$ and the less the work of the turbine stage:

$$L_u = \frac{u}{g}(c_1 \cos \alpha_1 \pm c_2 \cos \alpha_2). \qquad (2.12)$$

Consequently, at the same value of T_3^* the pressure drop π_{T}^{*} will be less. But in this case for maintaining equilibrium conditions of revolutions the fuel feed control will increase the fuel feed to the combustion chamber. As a result T_3^* will increase. This will lead to increase of π_{k}^{*} and to shift of the performance point of the turbocompressor to the region of raised values of T_3^* and π_{k}^{*}.

Thus, increase of the sections of jet nozzle and nozzle box of the turbine when n_{np}=const has the opposite effect on the change of π_K^* and T_3^*.

On Fig. 2.4 there is shown the family of lines of operating conditions when f_{CA} = const.

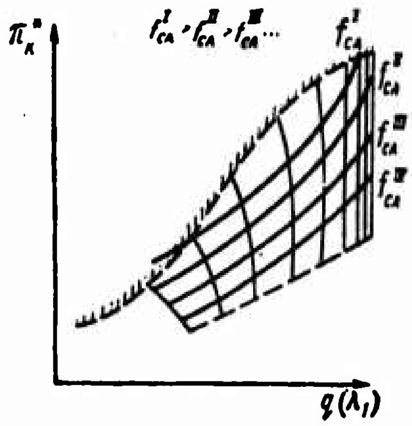

Fig. 2.4. Family of lines of operating conditions when F_{CA} = const.

The above-described method of the influence of the throat of the turbine nozzle box on engine parameters is frequently used in the special design office of aviation plants in the process of finalizing the experimental engines.

The variable-area nozzle boxes did not receive wide application on series-produced engines because of organic defects inherent to them: large gas leakages in the radial clearance and difficulties in providing reliable operation of the turning mechanism at high gas temperatures.

§ 2. Revolution Characteristics of TRD

Change of Efficiencies and Loss Factors of Basic Elements (Subassemblies) of TRD with Respect to the Number of Revolutions

Hydraulic and gas-dynamic perfection of the TRD is evaluated:

with the aid of efficiencies η^* of compressor and turbine;

coefficients of total pressure drop (loss factors) σ^* of intake, combustion chamber, afterburner, diffuser;

jet nozzle efficiency φ.

In the process of engine operation the efficiency of the compressor is changed most of all.

During engine throttling it increases at first to value 0.84 to 0.88, and then drops with further lowering of revolutions (Fig. 2.5).

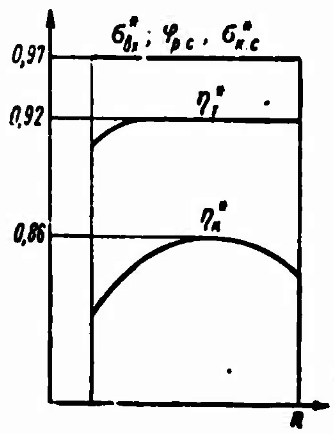

Fig. 2.5. Change of efficiency and loss factors of the basic elements of the TRD with respect to the number of revolutions.

Turbine efficiency has constant value 0.90-0.93 in a wide range of revolutions and drops only in the region of rough throttling.

Loss factors of the intake $\sigma^*_{вх}$, combustion chamber $\sigma^*_{к.с}$, diffuser $\sigma^*_{д}$, and also the jet nozzle efficiency $\varphi_{р.с}$ keep a constant value, equal to approximately 0.96-0.98, in the entire range of working revolutions.

Change of Gas Pressure in Characteristic Sections of the Gas-Air Duct of the TRD During Engine Throttling

With an increase of the number of revolutions of the TRD rotor the work and compression ratio of the compressor continuously increase. This follows from equation

$$L_к = 102{,}5\,T_0\,(\pi_к^{*0{,}286} - 1)\,\frac{1}{\eta^*_{к}} \sim n^2.$$

Consequently, the total air pressure behind the compressor p_2^* is continuously increased. This, in turn, leads to increase of the total gas pressure before the turbine p_3^*, at the turbine exhaust p_4^*, and also at the jet nozzle edge p_5^*.

Total pressure at the compressor inlet p_1^* is somewhat decreased with rise of the number of revolutions of the rotor as a result of increase of hydraulic losses at the engine intake with rise of inlet flow velocities of the engine, i.e., increase of $\lambda_{1a}(M_{1a})$.

Increase of the compression ratio of the compressor leads to the growth of expansion ratios of the turbine and jet nozzle, since

$$\pi_к^* \sigma_{вх}^* \sigma_{к.с}^* = \pi_т^* \pi_{р.с} \qquad \text{(Fig. 2.6)}$$

However, this process is accomplished only in the subsonic region of gas outflow from the jet nozzle up to the number of revolutions at which the pressure drop in the jet nozzle reaches critical value, i.e.,

$$\frac{p_4^*}{p_5 = p_H} = \pi_{кр} = \left(\frac{k+1}{2}\right)^{\frac{k}{k-1}} = 1{,}85 \text{ (when } k = 1.33).$$

With further rise of revolutions and, consequently, total expansion ratio the pressure drop p_4^*/p_5 in the jet nozzle remains constant no matter how the nozzle inlet pressure is increased. The

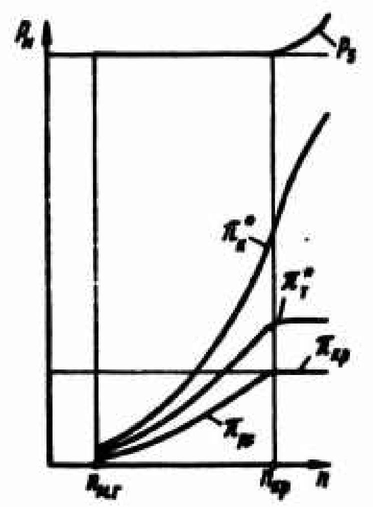

Fig. 2.6. Change of compression (expansion) ratios in elements of the TRD.

given circumstance leads to the fact that gas expansion in a usual convergent TRD nozzle already turns out to be incomplete. With increase of revolutions above "critical" at the nozzle edge the excess static pressure, i.e., $p_5 > p_H$, appears and continuously grows (Fig. 2.6).

The appearance of a critical pressure drop at the jet nozzle edge leads to "cutoff" of the turbine according to the pressure drop: expansion ratio of gas in the turbine remains constant no matter how the revolutions increase:

$$\overset{*}{\pi}_T = \frac{\overset{*}{p}_3}{\overset{*}{p}_4} = \text{const}.$$

This ensues from the transformed equation of flow rate, formulated for the "turbine-jet nozzle" system.

$$\overset{*}{\pi}_T^{\frac{n+1}{2n}} = \frac{f_5 q(\lambda_5)}{f_{CA} q(\lambda_{CA})}. \qquad (2.13)$$

Indeed, when $\lambda_{CA} = 1$ (or $\lambda_{CA} = \text{const}$), $n = \text{const}$ and at fixed geometry of the engine the value of $\overset{*}{\pi}_T$ grows with increase of λ_5 to 1; when $\lambda_5 = 1 = \text{const}$ we find that $\overset{*}{\pi}_T = \text{const}$.

Let us examine how the total expansion ratio of gas is redistributed between the turbine and jet nozzle of the TRD under subcritical conditions of outflow from the engine ($\lambda_5 < 1$).

Let us introduce the following assumptions:

1) $q(\lambda_{CA}) = 1$ or $q(\lambda_{CA}) = \text{const}$ in a wide range of throttle conditions;

2) $\overset{*}{\eta}_T = 1$ and, consequently, $n = \kappa$ (where n — polytropic exponent);

3) $\overset{*}{\sigma}_{p.c} = 1$, i.e., $\overset{*}{p}_5 = \overset{*}{p}_4$.

By using the system of relative parameters, let us represent equation of flow rate (2.13) in the form

$$\bar{\pi}_{\tau}^{\overset{\bullet}{}\frac{k+1}{2k}}=\bar{q}(\lambda_5). \qquad (2.14)$$

In equality

$$\bar{\pi}_{\kappa}^{\bullet}=\bar{\pi}_{\tau}^{\bullet}\bar{\pi}_{p.c}^{\bullet}$$

let us replace parameter $\bar{\pi}_{\tau}^{\bullet}$ by its value from equation (2.14). We obtain

$$\bar{\pi}_{\kappa}^{\bullet}=\frac{\bar{q}(\lambda_5)^{\frac{2k}{k+1}}}{\bar{\Pi}(\lambda_5)}, \qquad (2.15)$$

where

$$\bar{\pi}_{\kappa}^{\bullet}=\frac{\overset{\bullet}{\pi}_{\kappa}}{\overset{\bullet}{\pi}_{\kappa(\kappa p)}};\quad \bar{\Pi}(\lambda_5)=\frac{\Pi(\lambda_5)}{\Pi(\lambda_5)_{\kappa p}};$$

$$\bar{q}(\lambda_5)=\frac{q(\lambda_5)}{q(\lambda_5)_{\kappa p}};\quad \Pi(\lambda_5)=\frac{1}{\pi_{p.c}}=\frac{p_5}{\overset{\bullet}{p}_5}=\frac{p_H}{\overset{\bullet}{p}_4}.$$

Subscript «кр» pertains to critical outflow conditions. Let us introduce the gas-dynamic function of pressure redistribution

$$\mu(\lambda)=\frac{q(\lambda_5)^{\frac{2k}{k+1}}}{\Pi(\lambda_5)}. \qquad (2.16)$$

The dependence of $\mu(\lambda)$ on $\pi_{p.c}$ or $\Pi(\lambda)$ for k = 1.33 is shown on Fig. 2.7. With increase of λ_5 from 0 to 1 the function $\mu(\lambda_5)$ grows from 0 to

$$\mu(\lambda_5)_{\kappa p}=\frac{1}{\Pi(\lambda_5)_{\kappa p}}=1{,}85.$$

In this case the equation of pressure redistribution (2.16) will take such a form:

$$\mu(\lambda_5)=1{,}85\,\bar{\mu}(\lambda_5)=1{,}85\,\bar{\pi}_{\kappa}^{\bullet}=$$

$$=\frac{q(\lambda_5)^{\frac{2k}{k+1}}}{\Pi(\lambda_5)}. \qquad (2.17)$$

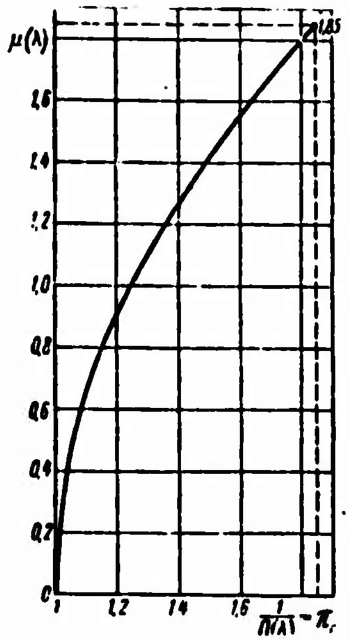

Fig. 2.7. Dependence of function $\mu(\lambda)$ on $\pi_{p.c}$.

Determination of π^*_T and $\pi_{p.c}$ at any value of $\pi^*_K < \pi^*_{K(kp)}$ is produced graphically or by tabular method according to the following scheme:

$$\pi^*_K \to \bar{\pi}^*_K = \bar{\mu}(\lambda_3) \to \mu(\lambda_3) \to \Pi(\lambda_3) \to \pi_{p.c} \to \pi^*_T.$$

Change of Gas Temperature in Characteristic Sections of the Gas-Air Duct with Respect to the Number of Revolutions

Stagnation temperature of air at the compressor inlet maintains a constant value, equal to the external air temperature at all revoluations:

$$T^*_1 = T^*_H = T_0 \text{ (when } H = 0 \text{ and } M_H = 0).$$

Air temperature past the compressor with increase of revolutions grows continuously approximately according to square-law variation. This may be seen from the equation of flow energy for the compressor

$$T^*_2 = T_0 + \frac{L_K}{102.5} = T_0 + An^2.$$

The most important and complicated function is the change of gas temperature before the turbine T^*_3 with respect to the number of

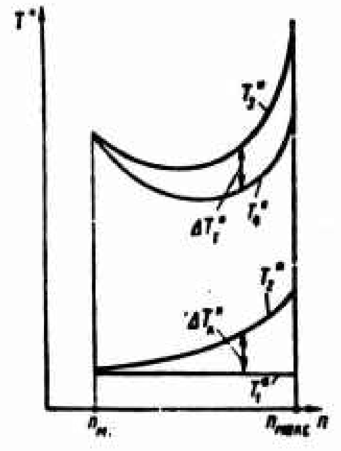

Fig. 2.8. Change of gas temperature before the turbine with respect to the number of revolutions of TRD.

revolutions (Fig. 2.8). On the one hand, the effectiveness of the engine cycle and its "outlet" indices depend on the value of this temperature, and on the other hand, the calorific intensity of the "hot" part of the engine. Change of the gas temperature before the turbine, depending on the number of revolutions, is determined by the equation of work balance of the turbocompressor

$$L_T = \frac{L_K}{\beta} \text{ or } 118 T_3^* e_T^* \eta_T^* = \frac{Cn^2}{\beta},$$

whence

$$T_3^* = \frac{Cn^2}{118 \beta e_T^* \eta_T^*}, \qquad (2.18)$$

where

$$e_T^* = 1 - \frac{1}{\pi_T^{*0,25}}.$$

In the subcritical region of gas outflow from the jet nozzle ($\lambda_5 > 1$), when $n > n_{кр}$, the pressure drop in the turbine π_T^* maintains a constant value. Consequently, in this region with decrease of revolutions the gas temperature before the turbine is lowered; it is approximately proportional to the square of the number of revolutions (see Fig. 2.8).

With further decrease of revolutions ($n < n_{кр}$) the turbine efficiency determined by expansion ratio π_T^*, starts to decrease. This leads to cessation of the temperature drop of gas T_3^*.

Under conditions of rough throttling of the engine, an intense drop of compressor and turbine efficiency sets in; in this case the expansion ratio of gas in the turbine is so insignificant that realization of equilibrium engine operating conditions (i.e., maintaining n = const) turns out to be possible only at raised values of gas temperature before the turbine; inasmuch as the drop of π_T^*, η_T^* and η_K^* is intensified with decrease of engine revolutions, increase of T_3^* also grows.

Thus, change of temperature T_3^* according to the revolutions is depicted by a concave curve with three characteristic segments (Fig. 2.9). With decrease of TRD revolutions the gas temperature before the turbine is at first sharply decreased (segment 1-2), then its drop is delayed in a wide range of revolutions and is practically ceased (segment 2-3) and, finally, in region of revolutions close to idling there occurs intense growth of T_3^* (segment 3-4). This "excess" of temperature is very considerable and sharp. Prolonged engine operation at idling can lead to impermissible overheating of the TRD and even to its failure. Therefore, it is necessary to have means and devices for preventing the undesirable increase of temperature T_3 in the zone of maximum and minimum engine revolutions.

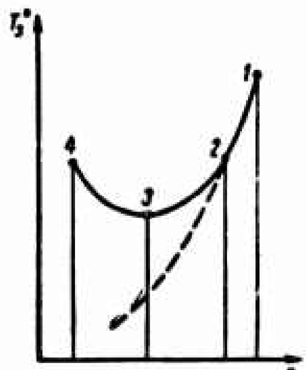

Fig. 2.9. Change of gas temperature T_3^* according to the number of revolutions.

With a short TRD exhaust duct it is possible to disregard heat removal through the jet nozzle wall. In this case the ram temperature of gas in the duct past the turbine (between segments 4-4 and 5-5) maintains constant value, i.e., $T_4^* = T_5^*$. The regularity of change of this temperature depending on the number of revolutions is similar to curve $T_3^* = f(n)$. Really, from the equation of flow energy for the turbine it follows that

$$T_4^*=T_3^*-\frac{L_T=L_K}{118}=T_3^*-Cn^2. \qquad (2.19)$$

Therefore, in the region of maximum and minimum engine revolutions in accordance with change of T_3^* there occurs increase of temperature T_4^*. However, as the revolutions are lowered the temperature range between curves T_3^* and T_4^*, proportional to the work of compressor L_K, is gradually decreased. In the zone of idling revolutions curves T_3 and T_4^* practically coincide. Thus, temperature T_4^* is a "companion" of temperature T_3^*. By deflection of temperature T_4^* from norm it is possible with good substantiation to judge the level of gas temperature before the turbine and the calorific intensity of the engine on the whole.

Change of Air Flow Rate According to the Number of Revolutions

From the theory of vaned machines it is known that the flow rate of air through the compressor increases approximately in proportion to its number of revolutions. However, in the region of maximum revolutions this relation, as experiment shows, is deflected from linear toward decrease of the flow rate of air (Fig. 2.10). Such regularity is connected with "blocking" phenomenon at the compressor inlet.

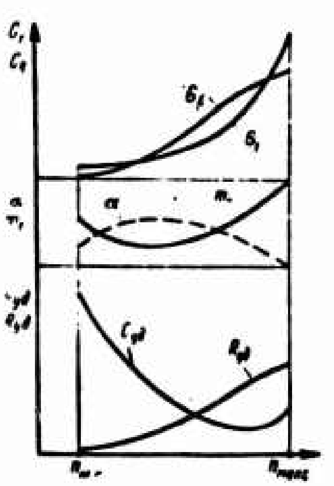

Fig. 2.10. Change of TRD parameters according to the number of revolutions.

As the axial flow rate of air c_{1a} approaches the speed of sound ($\lambda_{1a}\rightarrow 1$) the relative flow density $q(\lambda_1)$ approaches its limiting value

and, consequently, the rise of air flow rate is ceased, since

$$G = m\frac{p_1^*}{\sqrt{T_1^*}} f_1 q(\lambda_1) \sim q(\lambda_1).$$

Therefore, the closer the number λ_1 is to 1 at calculated conditions (which is caused by the tendency to decrease the overall diameter of the engine), the more the relationship G = f(n) is deflected from linear.

Change of Specific Thrust of TRD According to the Number of Revolutions

With increase of the number of revolutions the exit velocity of gases from the jet nozzle rises, equal to

$$c_5 = \varphi\sqrt{2g\frac{c_p}{A}T_4^* e_{p.c}},$$

where

$$e_{p.c} = 1 - \frac{1}{\pi_{p.c}^{\frac{k-1}{k}}}; \quad \pi_{p.c} = \frac{p_4^*}{p_H}.$$

Increase of c_5 is caused by growth of pressure drop in the jet nozzle, and in the region of maximum revolutions, furthermore, by increase of stagnation temperature of gas at the jet nozzle inlet.

Thus, with increase of the number of revolutions the specific thrust of TRD continuously increases (Fig. 2.10); it is equal to

$$R_{y\partial} = \frac{c_5}{g}.$$

Incomplete expansion of gas, occurring at supercritical pressure drops in usual (convergent) TRD nozzle, lowers the specific thrust of the engine. However, calculations show that at usual values of T_3^* and π_κ^* the difference between $R_{y\partial}$ at total expansion and under expansion of gas at maximum stand conditions does not exceed 2-3%.

Increase of specific thrust of TRD with rise of the number of revolutions can be given by another explanation, connected with change of work of the TRD cycle. It is known that when $T_3^* = \text{const}$ the value of L_e at first grows with increase of compression ratio. Growth of L_e is intensified at maximum revolutions, when intense increase of gas temperature before the turbine sets in. Increase of cycle work leads to a rise of exit velocity and, consequently, specific thrust.

Change of Total Thrust of TRD According to the Number of Revolutions

Change of total thrust of TRD according to the number of revolutions (Fig. 2.11) is completely determined by the regularities of change of specific thrust and the flow rate of air per second.

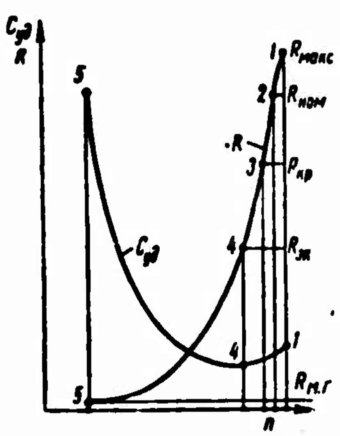

Fig. 2.11. Change of thrust and specific fuel consumption of TRD according to the number of revolutions.

Let us examine a TRD with average compression ratio of the compressor and a moderate value of axial velocity at the compressor inlet at maximum conditions ($\lambda_{1\alpha} = 0.6$). For these conditions we can sufficiently exactly consider that the flow rate of air is proportional to revolutions in the first power, and specific thrust is approximately proportional to the number of revolutions squared, i.e.,

$$G_в = C_1 n, \quad R_{уд} = C_2 n^2.$$

Then in the first approximation we obtain

$$R = R_{уд} G_в = C_3 n^3. \qquad (2.20)$$

Treatment of numerous experimental data shows that this dependence of thrust on the number of revolutions is valid for the majority of turbojet engines with program of control f_5 = const in the region of revolutions less than nominal. In the region of nominal and maximum numbers of revolutions the growth of TRD thrust is delayed in accordance with the regularity of rise of the flow rate of air (in this case the exponent at n is lowered).

Thus, the dependence of thrust on the number of revolutions is described by exponential dependence

$$R = An^x \text{ (where } A = \text{const)},$$

for which superscript x has variable value (x = 1-4).

Change of Total Excess Air Ratio According to the
Number of Revolutions

Let us first examine how the relative fuel consumption is changed:

$$m_т = \frac{c_{p_m}(T_3^* - T_2^*)}{\xi_{к.с} H_u} = \frac{G_т}{G_в}$$

according to number of revolutions of turbojet engine.

From equality $L_т = L_к$ with the assumption that $c_{pг} = c_{pв} = c_p$ follows approximate relationship

$$T_2^* - T_0 = T_3^* - T_4^* \text{ or } T_3^* - T_2^* = T_4^* - T_0.$$

Thus, parameter $m_т$ is changed in proportion to the stagnation temperature of gas past the turbine (see Fig. 2.10).

Total excess air ratio

$$\alpha = \frac{1}{m_т l_0}$$

is changed inversely proportional to the value of $m_т$. With decrease of revolutions at first there occurs leaning of the mixture (to α = 4.5-5); with further engine throttling the fuel-air mixture is enriched.

Change of Specific Fuel Consumption According to the Number of Revolutions

Change of specific fuel consumption according to the number of revolutions is determined by peculiarities of the change of parameters of the engine working process $\pi^*_к$, T^*_3 and $\eta^*_{к}$.

With decrease of engine revolutions from maximum to minimum the compression ratio of the compressor is lowered many times (from $\pi^*_{к(расч)}$ to a value approximately equal to 1). This causes increase of the specific fuel consumption during engine throttling. However, in the region of maximum revolutions, in which the absolute values of $\pi^*_к$ are still high, intense lowering of the gas temperature before the turbine, proportional to revolutions squared, and also some increase of compressor efficiency along the line of operating conditions lead to the fact that the specific fuel consumption of TRD is somewhat lowered at first (by 3-5%).

Thus, dependence of specific fuel consumption on the number of revolutions has the form of a concave curve sometimes with a clearly designated minimum, determining the minimum specific fuel consumption of TRD (see Fig. 2.11).

The obtained regularity can also be explained by the combined influence of two factors: specific thrust and relative fuel consumption, since

$$C_{уд} = 3600 \frac{m_т}{R_{уд}}.$$

Actually, during throttling the specific thrust of TRD drops continuously. The relative fuel consumption is changed in accordance with relationship $T^*_4 = f(n)$.

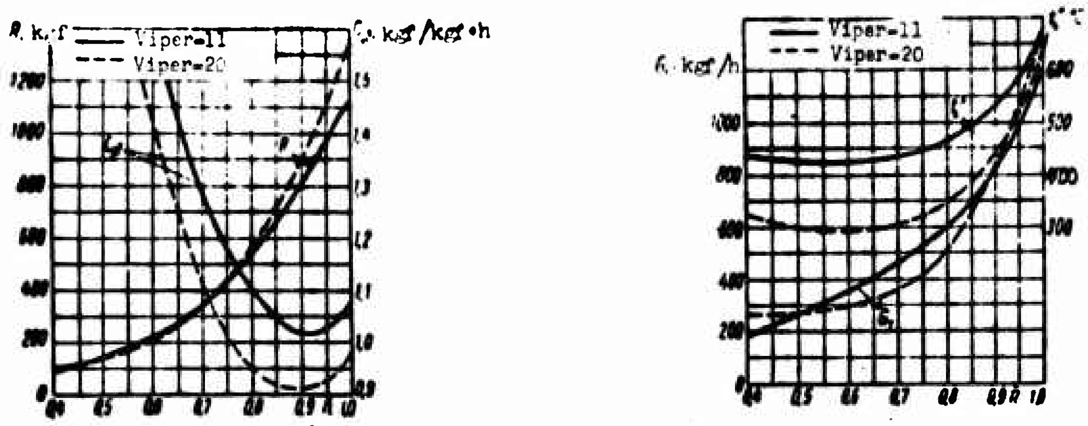

Fig. 2.12. Throttle characteristic of turbojet engines Viper-11 and Viper-20.

Initial leaning of the mixture predetermines the appearance of minimum on curve $C_{y\pi}$. Subsequent enrichment of the mixture (increase of m_T) intensifies the growth of $C_{y\pi}$ in the region of lowered TRD performance.

On Fig. 2.12 there are given throttle characteristics of English turbojet engines Viper-11 and Viper-20.

§ 3. Special Cases of Throttle Characteristics

Throttle Characteristics of Turbojet Engines at Constant Revolutions

When the turbojet engine is equipped with a variable area jet nozzle, starting of the engine and bringing it to maximum revolutions in a number of cases are accomplished at maximum nozzle opening. This substantially lowers the time of engine spin-up, prevents the appearance of compressor surging, and improves its pickup. After the TRD reaches maximum revolutions the engine thrust is augmented by a jet nozzle cover when n = const; in this case the compression ratio of the compressor and the gas temperature before the turbine increase. On the compressor characteristic (Fig. 2.13) there is depicted the line of operating conditi .ıs (LRR) 1-2-3 for the shown case. Simultaneous increase of the values of π_κ^* and T_3^* shar ly increases the specific thrust of turbojet engines. Regardin the air flow rate however, its change when n = const is determined by the

the peculiarity of the passage of the compressor pressure characteristic. Usually the air flow rate with covering of the nozzle is somewhat reduced (slanted characteristic) or remains constant (a case of vertical characteristic).[1]

Thus, engine thrust with TRD jet nozzle cover continuously increases. Specific fuel consumption in this case is at first lowered, and in the region of high values of T_3^* again increases.

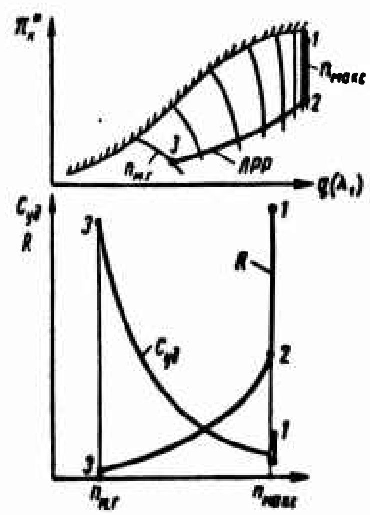

Fig. 2.13. Throttle characteristic of TRD when n = const.

Throttle Characteristic of TRD, Equipped with Compressor Air-Bleed Unit

Let us designate the number of revolutions, at which when throttling the TRD the compressor bleed unit is actuated, by $n_{пер}$. When $n<n_{пер}$ we have

$$\frac{G_г}{G_в}=\beta<1.$$

[1]The connection between $\pi_к^*$ and T_3^* with vertical compressor characteristic is determined by flow rate equation (2.8), from which we obtain

$$\pi_к^*\sim\sqrt{T_3^*}.$$

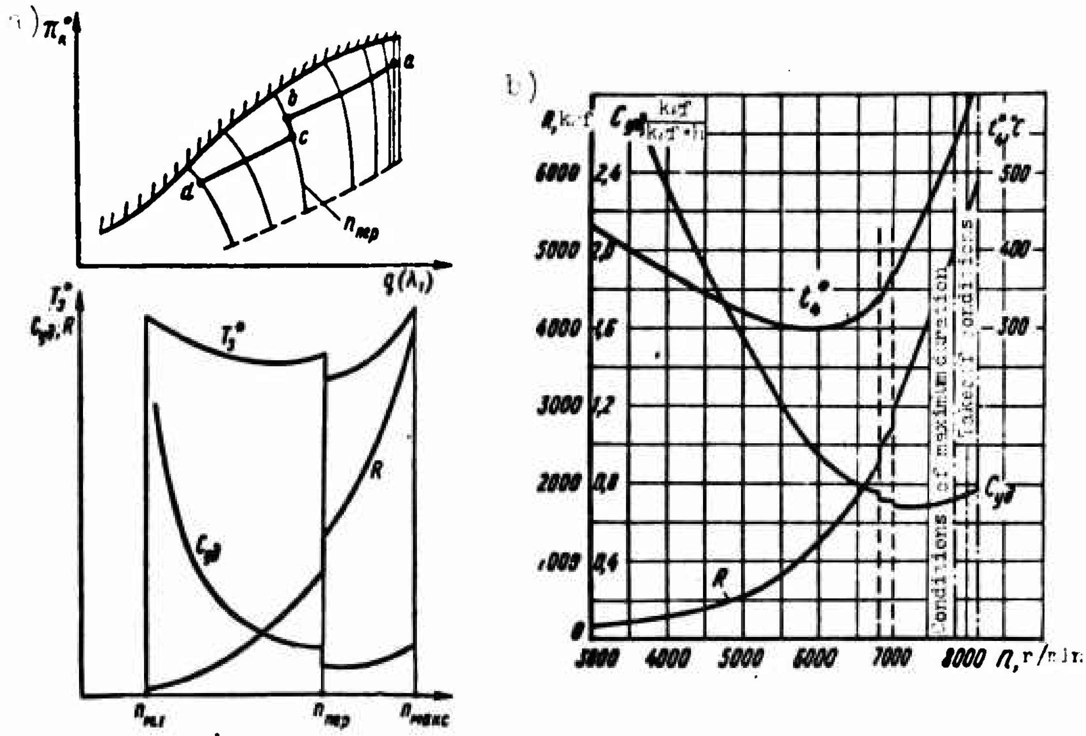

Fig. 2.14. Throttle characteristic of TRD with one bleed valve (a) with two bleed valves — engine "Avon" (b).

On Fig. 2.14 there are given throttle characteristics of a TRD equipped with compressor air-bleed unit, and also a compressor characteristic with LRR a-b-c-d plotted on it. During engine throttling in the region of revolutions $n_{пер}<n<n_{макс}$ (when bypass valves are closed) curves $R=f(n)$, $C_{уд}=f(n)$ and $T_3^* = f(n)$ do not differ at all from earlier examined dependences.

When $n=n_{пер}$ there occurs exhaust of part of the air into the external medium. The disturbance due to this material balance ($G_г<G_в$ and $\beta<1$) leads accordingly to disturbance of energy balance ($N_т<N_к$). For maintaining equilibrium operating conditions of the turbocompressor the speed regulator increases the fuel feed to the combustion chamber and the gas temperature before the turbine increases to the necessary degree.

Since with opening of bypass valves the backpressure at the exit of first the compressor stages is lowered, the flow rate of air through the compressor usually increases somewhat. As a result the

compression ratio $\pi_к^*$ drops (curve b-c). The pressure along the entire gas-air duct is lowered, accordingly. As a result the exit velocity of gas, specific thrust, and the flow rate of gas through the jet nozzle are decreased. Consequently, total engine thrust drops. Specific fuel consumption of the TRD grows. The latter is explained by the fact that the interval of gas preheating in the combustion chamber $(T_3^* - T_2^*)$, is increased, whereas specific thrust is substantially lowered.

The growth of $C_{уд}$ is explained physically by drop of effective cycle efficiency because of the worst use of heat at lowered pressure of the working medium, and also the uneconomical use of compressed air.

Throttle Characteristic of TRD, Equipped with Rotary Compressor Stator

Let us assume that when throttling the engine at a certain number of revolutions $(n_{НА}<n_{макс})$ the blades of the rotary stator turn at angle $\Delta\varphi$ in the direction of rotation of the compressor and further decrease of engine revolutions is produced at the new and constant position of the blades (for example, $\varphi_{НА}=10°$).

As a result of rotation of the stator in the direction of rotation of the rotor the work of the compressor and its compression ratio are decreased. Consequently (assuming in the region of high revolutions that $\pi_т^*=\text{const}$), the gas temperatures before the turbine T_3^* and past it T_4^* are lowered [see equation (2.18)], air flow rate, specific and total thrust drop, specific fuel consumption is also lowered, the surge margin increases. With high throttling of TRD a relative increase of $C_{уд}$ and T_4^* is possible due to severe lowering of $\pi_к^*$ and $\pi_т^*$ (as compared to a case when the stator is fixed).

A segment of TRD characteristic with stator blades turned (at angle $\varphi_{НА}>0°$) is shown on Fig. 2.15 in the form of dot-dash curves.

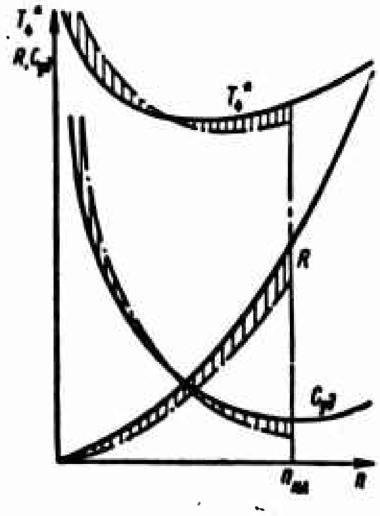

Fig. 2.15. Throttle characteristic of TRD, equipped with rotary compressor stator (PNA). Solid curve when $\varphi_{HA}=0°$, dot-dash when $\varphi_{HA}>0°$.

Region of Possible Operating Conditions of a Turbojet Engine

It was shown above that with adjustment of the jet nozzle exit section the line of operating conditions of TRD, plotted on the compressor characteristic, is displaced and thus describes a certain region of possible operating conditions.

However, operation of the turbojet engine does not turn out to be physically possible and permissible in all of this region. At certain engine operating conditions in its separate elements (compressor, combustion chamber, turbine) there appear disturbances and changes, which limit and narrow the actual region of possible operating conditions. Certain TRD operating conditions cannot be physically realized.

These limitations are connected with the appearance of:

unstable operation in the engine elements;

gas-dynamic "choking" of separate sections of the gas-air duct;

danger of disturbance of the durability and reliability of engine operation and its separate subassemblies (for example, with increase of engine revolutions and gas temperature before the turbine above permissible limits).

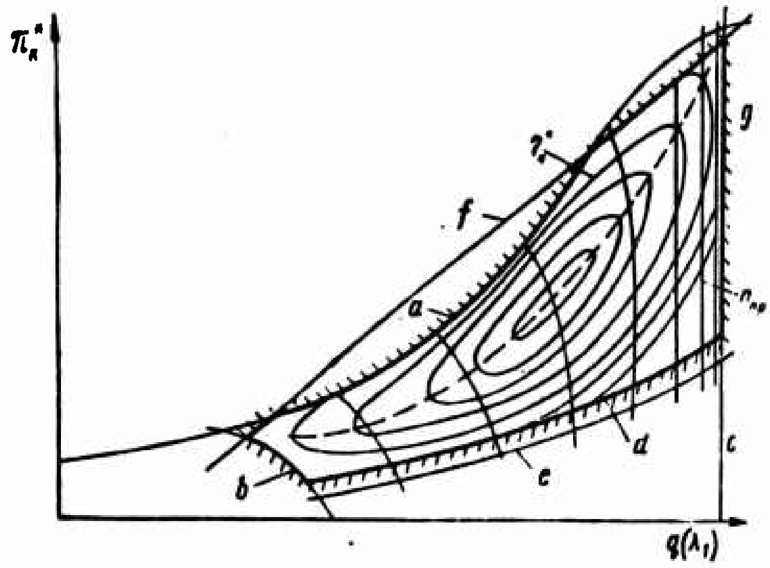

Fig. 2.16. Typical compressor characteristic with lines of limitation of TRD operating conditions plotted on it.

On Fig. 2.16 there is presented a typical compressor characteristic with limitation lines of TRD operating conditions plotted on it.

Here a – boundary of stable compressor operation;

b – boundary of stable combustion chamber operation (at low revolutions);

c – line of gas-dynamic "choking" at compressor inlet; theoretically, "choking" sets in when $q(\lambda_1) = 1$, practically (taking into account the convergence in the inlet channel) when $q(\lambda_1) \approx 0{,}80$–$0{,}85$;

d – line of "choking" at compressor outlet; "choking" sets in with approach of number λ at compressor outlet ($\lambda_{2\alpha}$) to 1 [$q(\lambda_{2\alpha}) = 1$];

e – line of "choking" at turbine exhaust; theoretically, "choking" sets in when $q(\lambda_{4\alpha}) = 1$, practically, (taking into account convergence in the channel past the turbine) when $q(\lambda_{4\alpha}) = 0.70$-$0.75$;

f – line of maximum permissible gas temperature before the turbine;

g – line of maximum permissible revolutions.

Peculiarities of Throttle Characteristics of Two-Shaft TRD

Throttle characteristics of two-shaft TRD are depicted in the form of dependences of basic engine parameters on the number of revolutions of the rotor whose rotation regulator is connected with the automatic combustion chamber fuel feed device.

Let us assume that fuel feed is connected with the rotation regulator of high-pressure rotor. With decrease of fuel feed to the combustion chamber the gas temperature before the turbine T_3^* is lowered. Due to this work of the high-pressure turbine drops. The advanced unbalance of works $L_{ТВД} < L_{КВД}$ is overcome by lowering revolutions of the high-pressure rotor. As a result for the latter the total compression ratio drops, and thus the total expansion ratio of gas. This is reflected first of all on the pressure drop of the low-pressure turbine, which is lowered considerably more intensely than for a high-pressure turbine.[1] Since when throttling the TRD by the number of revolutions the low-pressure compressor is "loaded" (blade angles of attack increase, which leads to relative increase of $\pi^*_{КНД}$ and $L_{КНД}$), and the high-pressure turbine is "lightened" (turbine work drops because of lowering of $\pi^*_{ТНД}$ and T_4^*), the unbalance of works $L_{ТНД} < L_{КНД}$ appearing now is eliminated due to more intense drop of revolutions of the low-pressure rotor than the high-pressure rotor.

As a result there sets in phenomenon called <u>slip</u> of rotors, when the ratio $\frac{n_{НД}}{n_{ВД}}$ is decreased more intense the deeper the conditions of throttling.

Ratio $\frac{n_{НД}}{n_{ВД}}$ increases with increase of fuel feed.

As it is known, slip of rotors gives the possibility of automatically improving the antisurge properties of the engine.

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[1]Drop of $\pi^*_{ТВД}$ sets in only when the first nozzle box of the low-pressure turbine passes into subcritical operating conditions.

Peculiarities of the Passage of Lines of Operating
Conditions of High-Pressure and Low-Pressure
Compressors of Single-Shaft and
Two-Shaft Turbojet Engines

Let us examine the peculiarities of passage of lines of operating conditions of high-pressure and low-pressure compressors of single-shaft and two-shaft turbojet engines.

Let us assume that low-pressure and high-pressure stages, consisting of compressor and turbine each, installed on one shaft, are assigned. Let us assume that the geometry and characteristics of the compressors and turbines, forming the stages, are known.

First we compare lines of operating conditions of high- and low-pressure turbocompressors in a two-shaft TRD (with free high-pressure and low-pressure stages) and in single-shaft TRD (with these rigidly connected stages, Fig. 2.17a).

Let us assume that at initial, calculated conditions the parameters and output indices of single-shaft and two-shaft turbojet engines are identical. In this case the calculated operating conditions of high-pressure and low-pressure compressors are depicted on the characteristics of corresponding compressors of single-shaft and two-shaft turbojet engines by the same point O.

During throttling of fuel feed [illegible] combustion chambers line OB of the operating conditions of low-pressure compressor of single-shaft TRD will be mildly sloping, approaching the surge boundary; this is explained by increase of blade angles of attack of the first compressor stages (i.e., low-pressure stage). Line of operating conditions of the high-pressure compressor (OB') of single-shaft TRD will be steep, at low revolutions approaching intersection of the region of "choking" conditions (so-called turbine conditions).

Now on the low-pressure compressor characteristic let us plot the line of operating conditions of free stage (OA), working in the system of a two-shaft TRD. It will pass considerably steeper than

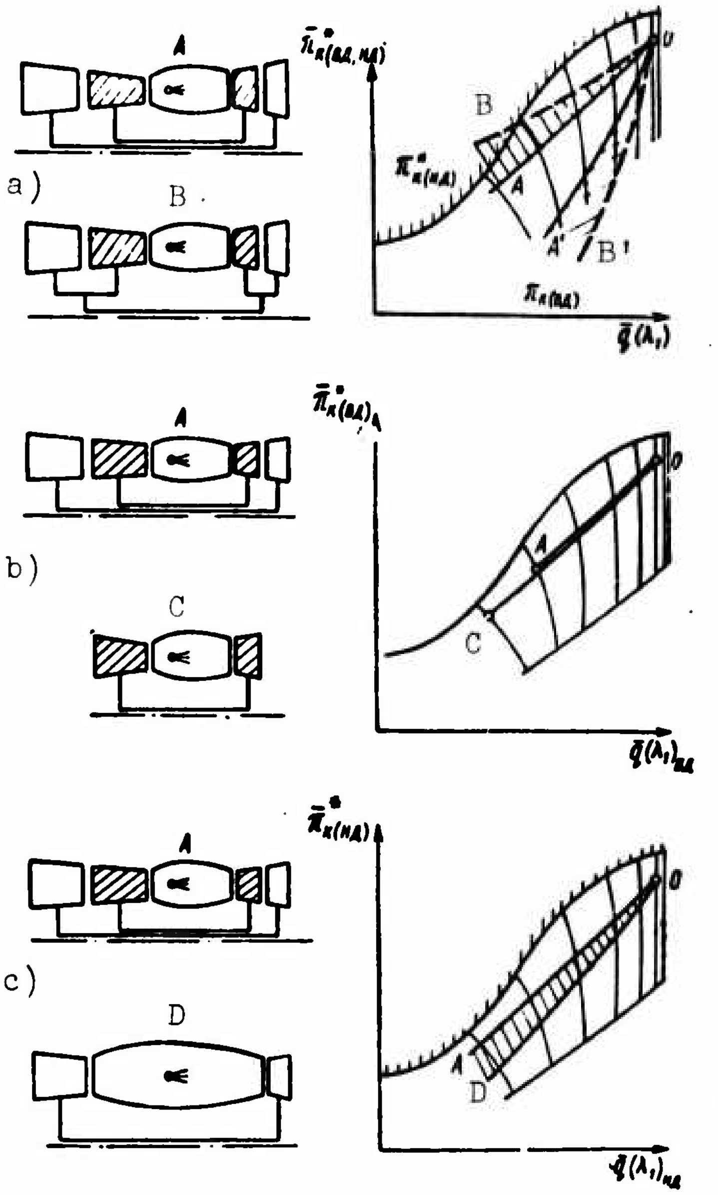


Fig. 2.17. Comparison of lines of operating conditions of turbocompressors of single-shaft and two-shaft TRD.

the line of operating conditions of the connected stage (OB). The line of operating conditions of the free high-pressure stage (OA'), conversely, will pass considerably flatter than the line of the connected high-pressure stage OB', so that "choking" conditions will be automatically removed.

It is possible to explain this regularity in the following way.

Let us assume that at a certain number of revolutions n_1 of single-shaft TRD the gas flow rate is equal to G_1. Mentally let us disengage the high- and low-pressure stages at this number of revolutions. Then the single-shaft TRD will be turned into a two-shaft TRD. As a result of the established unbalance of works of the low-pressure compressor and turbine $(L_{\text{КНД}} > L_{\text{ТНД}})$ the number of revolutions of the low-pressure compressor will drop and will become $n_{\text{НД}} < n_1$.

The established unbalance of works of high-pressure compressor and turbine $(L_{\text{КВД}} < L_{\text{ТВД}})$ will lead to increase of the number of revolutions of the high-pressure compressor, which will now be greater than n_1, i.e., $n_{\text{ВД}} > n_1$.

As a result of disconnection of stages the compression ratio of the low-pressure compressor will drop, and that of the high-pressure compressor will be increased. Total compression ratio, and also the flow rate of air through the TRD will practically not be changed.

It is not difficult to conclude from Fig. 2.17a that drop $\pi^*_{\text{КНД}}$ when G = const signifies the displacement of line of operating conditions toward freeing the low-pressure compressor from surge. Increase of $\pi^*_{\text{КВД}}$ when G = const leads to displacement of the line of operating conditions toward freeing the high-pressure compressor from "choking" conditions.

Elimination of surge for the low-pressure compressor when throttling the two-shaft TRD by the number of revolutions is physically explained by the fact that decrease of the peripheral velocity of blades at the same value of axial velocity leads to reduction of the angles of attack of the buckets; as a result of this the compression ratio of the stage drops and the danger of flow separation from the blade back of the compressor is reduced. Analogously, freeing the high-pressure compressor from the "choking" zone is explained by the fact that increase of peripheral velocity of blades at the same value of axial velocity leads to increase of angles of attack of the buckets; due to this, the compression ratio of the stage increases and the stages of the high-pressure compressor are removed from "choking" conditions.

If now on the compressor characteristic, constructed in relative parameters $\bar{\pi}_{\kappa}^{*}=f[\bar{q}(\lambda_{1})]$, we plot lines of operating conditions of high- and low-pressure compressors (see Fig. 2.17a), then the line of operating conditions of low-pressure compressor (OA) will pass considerably flatter (i.e., with a smaller stability margin) than this same line of the high-pressure compressor (OA'). In the region of revolutions higher than calculated the properties of line of operating conditions are changed: in it the line of operating conditions of high-pressure compressor occurs with a smaller stability margin than for the low-pressure compressor.

Let us now compare lines of operating conditions of free high-pressure stage in a two-shaft TRD and the same separately taken turbocompressor stage of a single-shaft TRD (see Fig. 2.17b).

At the conditions stipulated above the lines of operating conditions of the given high-pressure stages coincide. Really, the influence of the low-pressure compressor on the high-pressure compressor in a two-shaft TRD is reduced to change of the total pressure and stagnation temperature of air at the high-pressure compressor inlet. In other words, the low-pressure compressor fulfills the role of boost stage at the high-pressure compressor inlet. In this case the change of total pressure does not affect the position of lines of operating conditions. Change of stagnation temperature ($T^{*}_{1\text{вд}}$) changes the given number of revolutions of the high-pressure stage, therefore, at equal physical revolutions of the high-pressure compressor in the two examined cases the given compressor revolutions will not coincide, they will always be less for the high-pressure compressor stage, with boost stage at the inlet, i.e.,

$$n\sqrt{\frac{288}{T^{*}_{1\text{вд}}}}<n\sqrt{\frac{288}{T^{*}_{H}}}.$$

Let us assume that the initial operating conditions of these two stages coincide (point O, Fig. 2.17b). Then with equal degree of decrease of physical revolutions the given revolutions of KVD will be greater than for the compressor of a single-shaft TRD because of intense lowering of $T^{*}_{1\text{вд}}$, induced by "slip" of rotors, i.e., $n_{\text{пр(A)}} > n_{\text{пр(B)}}$.

In conclusion let us compare the lines of operating conditions of the free low-pressure stage of a two-shaft TRD and the same turbocompressor stage of single-shaft TRD taken separately (Fig. 2.17c).

Under the formulated conditions the lines of operating conditions of this stage do not coincide in the examined cases. At a general performance point under maximum conditions the line of operating conditions of the free cascade (OA) is more mildly sloping, i.e., with smaller stability margin than for the separately taken low-pressure compressor (OD)

The influence of high-pressure turbine (TVD) on the low-pressure turbine (TND) is reduced to additional change of gas parameters at the TND inlet. The high-pressure turbine should be considered as the expansion (throttling) stage at the low-pressure turbine inlet.

The above-mentioned regularity of the line of operating conditions is explained in the following way. When throttling a two-shaft TRD $\pi^*_{\kappa II}$ and T^*_3 are lowered. However, the pressure drops more intensively than the gas temperature, as a result of which the density of gas at the low-pressure turbine nozzle box inlet is sharply lowered, and this is equivalent to the effect of choking of the sections in the turbine or additional preheating of gas in the combustion chamber. As a result the line of operating conditions of the free low-pressure stage is deflected to the region of raised values of π^*_κ.

§ 4. Basic Operating Conditions of TRD

At the present time there is still not established common nomenclature of the basic operating conditions of gas-turbine engines. Abroad every firm manufacturing aircraft engines, and every airline, utilizing these engines in the course of finishing and operating the gas-turbine engines improves and changes the list of basic engine operating conditions, the relationship between thrusts, and the value of basic engine parameters under these conditions. In the USSR for turbojet engines the following nomenclature of basic conditions is

accepted: maximum (or takeoff), nominal cruising, economical and idling.

Maximum (or Takeoff) Performance

Conditions at which the engine develops maximum thrust during continuous operation for a limited time, as a rule not more than 5-10 min, are called maximum (takeoff) performance. Under these conditions (maximum number of revolutions) the maximum permissible gas temperature past the turbine is limited from conditions of safeguard of reliable engine operation. Maximum performance is used on takeoff, during climb, for achievement of maximum flight speed under combat conditions (during pursuit of the enemy, departure from him).

Nominal Performance

Conditions at which the engine develops the most thrust during continuous operation for 30 min (up to 1 h) are called nominal performance. Under these conditions (nominal number of revolutions) the maximum permissible gas temperature after the turbine is also limited. Thrust at nominal performance is usually 10-15% lower than at maximum conditions:

$$R_{ном} \approx (0{,}85 - 0{,}90) R_{макс};$$

in this case $n_{ном} \approx (0{,}96 - 1) n_{макс}$.

Basic calculations of the engine (for strength, gas-dynamic, selection of flow passage cross sections) are produced for nominal performance.

Nominal performance is basic for operation of the engine on a fighter airplane. On passenger aircraft this performance is used during climb.

Cruise Performance

Conditions at which there is guaranteed the greatest thrust at continuous and reliable engine operation during the entire set service life (lifetime) are called cruise performance. It is used during maximum-range route flights.

Thrust at cruise performance is 25-30% lower than at maximum performance:

$$R_{кр}=(0,70-0,75)R_{макс}\approx 0,85R_{ном},$$

in this case $n_{кр}\approx 0,9n_{макс}$.

The shown conditions are frequently called maximum cruise.

Economical Performance

Conditions which approximately correspond to minimum specific fuel consumption are called economical performance. Thrust under the given conditions is 40-50% less than at maximum:

$$R_{эк}=(0,50-0,60)R_{макс}\simeq 0,70R_{ном}.$$

This performance is also called reduced cruise performance.

There are TRD operating conditions reduced still more (0.6 and 0.4 $R_{ном}$).

Idling

Conditions at which the engine operates stably at minimum revolutions for a limited time (10-15 min) are called idling. Thrust at the given conditions is 3-5% of maximum (at $R_{макс}=5000$ kgf we have $R_{м.г}=150-250$ kgf). This thrust should be sufficient for taxiing the aircraft on the airfield, but should not be excessive in avoidance of increase of the aircraft landing run when landing with engines operating.

Usually $n_{м.г} = (0,2-0,4)\, n_{макс}$.

At idling the gas temperature after the turbine is also limited.

For special extraordinary cases certain foreign firms, manufacturing aircraft engines, introduce extraordinary conditions. Under these conditions in case of an emergency situation the engine should operate 1-2 min without failure.

§ 5. The Influence of Atmospheric Conditions on TRD Operation

Above we examined the throttle characteristics of a TRD in detail, considering during their analysis that the atmospheric conditions remain constant. However, the last (temperature T_H and pressure p_H) essentially affect the parameters of the working process, operating conditions, and basic indices of the turbojet engine.

Let us explain this in more detail. Let us assume that the number of engine revolutions and the position of control elements remain constant (n = const and f_5 = const).

Let us first consider a case when the atmospheric pressure is increased. Increase of p_H, and thus the air density, cause increase of the mass flow rate of air through the engine. Since change of p_H causes a proportional change of pressure over the entire engine duct, the exit velocity of gas from the jet nozzle, and consequently, the specific thrust of TRD, are not changed. As a result the total thrust of TRD increases in proportion to increase of atmospheric pressure. With lowering of p_H, conversely, TRD thrust drops.

Let us consider now a case when the external temperature is reduced. Lowering of T_H causes increase of the mass flow rate of air through the engine (air density γ_H rises); furthermore, specific thrust of the TRD rises, since with the same expended work for compression of 1 kgf of air

$$L_к = 102,5 T_H (\pi_к^{*0,286} - 1) \frac{1}{\eta_к^*} = \text{const}$$

there are increased values of the compression ratio of compressor and as a result the gas pressure at the jet nozzle inlet. In the final analysis the exit velocity c_5 from the engine is increased. This leads to increase of TRD thrust as a result of change of its two components ($G_в$ and $R_{уд}$), and not one ($G_в$), as in the preceding case. Increase of T_H, conversely leads to drop of TRD thrust.

Relative oscillations of external pressure are considerably less than for the external temperature. Actually, the real change of temperature from -45°C in winter to +45°C in the summer relative to standard temperature t_H = 15°C composes value

$$\Delta\bar{T}=\frac{\Delta T}{T_{ст}}=\frac{90}{288}\simeq 31\%.$$

Change of p_H from 720 to 780 mm Hg relative to standard value of pressure p_H = 760 mm Hg gives only

$$\Delta\bar{p}=\frac{\Delta p}{p_{ст}}=\frac{60}{760}=8\%.$$

Therefore, change of external temperature has a greater effect on engine operation than change of external pressure. Thus, for example, during utilization of aircraft with TRD under conditions of the extreme north the engines develop substantially greater thrust (30-40%) than in the southern latitudes. Increase of ambient air temperature from +15 to +30°C (i.e., by 5%) leads to 7-11% drop of TRD thrust.

The influence of temperature T_H and pressure of ambient air p_H on the throttle characteristics of TRD is shown on Fig. 2.21.

Thus, change of ambient atmospheric conditions essentially affects TRD operation. The question appears, how can this influence on the throttle characteristics of the engine be considered? Stand characteristics of TRD are taken at different atmospheric conditions.

What engine indices should be recorded in its log book? How can we compare with each other the characteristics, for example, of

engines of the same series, taken at different values of p_H and T_H?

It would be senseless to try to take characteristics of TRD at exactly the same atmospheric conditions.

It is obvious that correct resolution of the problem of calculation of the effect of ambient atmospheric conditions on TRD characteristics consists of eliminating this influence, having taken certain atmospheric conditions as "standard," reducing the results of tests obtained at any conditions to these standard conditions and, consequently, constructing any throttle characteristic of TRD only for these conditions.

The following conditions are accepted as standard: $t_0 = +15^{\circ}C$ ($T_0 = 288^{\circ}K$), $p_0 = 760$ mm Hg.

Conversion of TRD characteristic, obtained at any external conditions, to standard is accomplished with the aid of the gas-dynamic similarity theory.

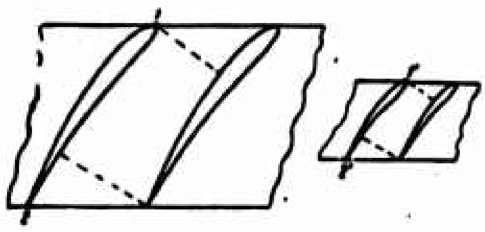
§ 6. Application of Gas-Dynamic Similarity Theory to Gas-Turbine Engines

The gas-dynamic similarity theory is discussed in detail in special courses of aerodynamics, thermogas-dynamics and vaned machines. Here we are limited to the discussion of basic concepts of gas-dynamic similarity, and also to consideration of the most important flow properties at similar conditions and conclusions, which can be used in reference to gas-turbine engines, in particular to turbojet engines.

Basic Information from the Theory of Gas-Dynamic Similarity

By gas-dynamic similarity we mean the similarity of gas flows. It signifies the similarity of physical fields of three parameters of flow: pressures, temperatures and velocities.

Geometric similarity signifies the similarity of the shape of flows, geometric contours of channels, air-gas flow area of the engine and its elements. In a number of cases, for example when examining the engine characteristics, when the same element of the engine operates under similar conditions, there is observed geometric identity of flows. Geometric similarity assumes the presence of similar points, lines, sections and volumes, similarity located in the examined systems (Fig. 2.18).



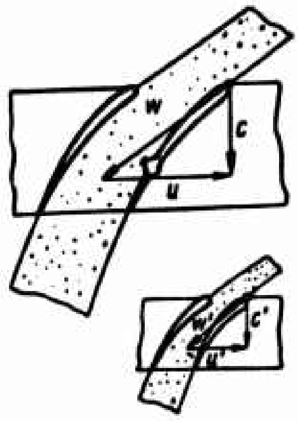


Fig. 2.18   Fig. 2.19

Fig. 2.18. Geometric similarity of aerodynamic grids.

Fig. 2.19. Kinematic similarity of flows in turbomachines.

Kinematic similarity signifies the similarity of velocity fields of the examined flows; it exists when at similar points of systems the gas velocities are parallel and are proportional to each other, i.e., their relationship is a constant:

$$\frac{c_2}{c_2'} = \frac{c_1}{c_1'} = \mathrm{const} \text{ or } \frac{c_2}{c_1} = \frac{c_2'}{c_1'} = \mathrm{const}.$$

With the presence of compound motion, when gas flows in revolving channels, the kinematic similarity signifies the similarity of velocity triangles at similar points or sections (Fig. 2.19). For this case it is possible to write the following equalities:

$$\frac{u}{c} = \frac{u'}{c'};\ \frac{w}{c} = \frac{w'}{c'}, \text{ etc.}$$

or

$$\frac{u}{u'} = \frac{c}{c'} = \frac{w}{w'},$$

where w, u, c — relative, movable (peripheral) and absolute gas velocities, respectively.

Dynamic similarity is characterized by proportionality of the forces affecting similar elements of flow. It signifies the similarity of fields of pressures, i.e., the constancy of relationship of pressure forces at similar points and sections of channels:

$$\frac{p_2}{p_1} = \frac{p_2'}{p_1'} \text{ or } \frac{p_2}{p_2'} = \frac{p_1}{p_1'} = \text{const.}$$

In energy isolated flows the similarity of pressure fields automatically leads to similarity of temperature fields, i.e., to constancy of relationship of temperatures at similar points.

The last similarity ensues from the adiabaticity of ideal flows or polytropicity of real flows (when n = const), i.e.,

$$\frac{T_2}{T_1} = \left(\frac{p_2}{p_1}\right)^{\frac{n-1}{n}}.$$

Thus we have

$$\frac{T_2}{T_1} = \frac{T_2'}{T_1'} \text{ or } \frac{T_2}{T_2'} = \frac{T_1}{T_1'} = \text{const.}$$

If, however, the flows are not energy isolated (there is supply or removal of heat), but there exists thermal similarity (similarity of temperature fields and heat flows), then the relationships of temperatures in the shown equalities are observed as before.

Similarity of fields of basic physical parameters of gas flow (p, T and c) determines the invariability of numbers M (or λ) in all flow sections under similar conditions.

Indeed, from similarity of velocity fields (kinematic similarity) and temperature fields (dynamic and thermal similarity) follows:

$$\frac{c_2}{c_1} = \frac{c_2'}{c_1'} = c_c = \text{const}, \tag{2.21}$$

$$\frac{T_2}{T_1} = \frac{T_2'}{T_1'} = c_T = \text{const}, \tag{2.22}$$

or

$$\sqrt{\frac{T_2}{T_1}} = \sqrt{c_T} = \text{const}. \tag{2.22'}$$

Having divided expression (2.21) by (2.22') we obtain

$$\frac{M_2}{M_1} = \frac{M_2'}{M_1'} = c_M = \text{const}. \tag{2.23}$$

In formulas (2.21), (2.22) and (2.23) c_c, c_T and c_M are complexes of similarity.

For sections 1 and 2 of flows let us now write equation of energy (assuming for simplicity that the examined flows are energy isolated)

$$c_p T_1\left(1 + \frac{k-1}{2} M_1^2\right) = c_p T_2\left(1 + \frac{k-1}{2} M_2^2\right) \tag{2.24}$$

and reduce it to form

$$\frac{T_2}{T_1} = \frac{1 + \frac{k-1}{2} M_1^2}{1 + \frac{k-1}{2} M_2^2} = \frac{1 + \frac{k-1}{2} M_1'^2}{1 + \frac{k-1}{2} M_2'^2} = c_T. \tag{2.25}$$

By comparing equations (2.23) and (2.25) with each other, it is possible to easily conclude that these equalities are possible only under the condition that

$$M_1 = \text{const}; \quad M_2 = \text{const}; \quad M_i = \text{const}$$

and

$$M_1 = M_1'; \quad M_2 = M_2'; \quad M_i = M_i'.$$

where i — any point of flow.

Thus, under similar conditions the M numbers in all sections of the channel keep a constant value, and are identical at similar points.

It is necessary to note that the above-mentioned properties of flows are valid with the presence of gas-dynamic similarity when observing the following preliminary conditions:

invariability of adiabatic index $k = \frac{c_p}{c_v} = \text{const}$;

constancy of Reynolds number (Re = const);

constancy of Prandtl number (Pr = const);

constancy of chemical composition of gas (absence of chemical reactions).

Deviation from these conditions in a number of cases (at high altitudes, at high supersonic flight speeds) predetermines the proximity of obtained relationships and regularities.

Properties of Similar Flows.

At similar conditions the relative and dimensionless parameters, which characterize flow, keep a constant value. Therefore, the efficiency of a gas-dynamic machine or unit, polytropic exponent of the process (n), compression and expansion ratios, relative thrust, air flow rate, fuel consumption, work and power are constant, i.e.,

$$\eta = \text{const}; \quad n = \text{const}; \quad \bar{R} = \frac{R}{R_{\text{макс}}} = \text{const};$$

$$\bar{N} = \frac{N}{N_{\text{макс}}} = \text{const}, \text{ etc.}$$

Characteristic Criteria of Gas-Dynamic Similarity.

Among the many criteria of similarity there exist those, keeping of constant values of which automatically determines the presence of gas-dynamic similarity.

The number of characteristic criteria of similarity usually corresponds to the number of independent motions. Thus, for a motionless channel with fixed cross sections there is one characteristic criterion — number M_a with respect to axial velocity. For a revolving channel or airfoil lattice with two independent forms of motion (relative and movable) there are two characteristic criteria: M number with respect to axial and peripheral velocity. Thus, conditions of providing gas-dynamic similarity for such channels are M_a = const and M_u = const.

Application of the Gas-Dynamic Similarity Theory to a Turbojet Engine

While considering the TRD as a totality of separate gas-dynamic elements — intake, compressor, combustion chamber, turbine and jet nozzle — the conclusion should be made that the gas-dynamic similarity of TRD on the whole assumes the observance of similarity of all its parts. However, it is possible to show that in certain elements of the engine, for example, for the intake and jet nozzle, the gas-dynamic similarity is rarely observed. In other elements of the TRD, for example in combustion chambers and afterburners, the realization of similar conditions turns out to be generally impossible. The gas-dynamic similarity theory is applicable mainly for vaned machines: compressor (fan) and turbine.

Thus, it is more correct to indicate not total similarity of TRD operating conditions (which does not exist in nature), but partial, meaning by it the similarity of operating conditions of its turbocompressor part.

Let us analyze this question in greater detail.

Intake.

Above it was shown that a necessary condition of gas-dynamic similarity is kinematic similarity, i.e., similarity of the configuration of flows, proportionality of velocities at similar points.

Let us examine the spectrum of streamlines of a standard intake during engine operation on a stand ($c_0 = 0$) and in flight ($c_0>0$).

On a stand the streamlines at the engine inlet converge in the form of a funnel, forming a natural convergent channel (Fig. 2.20a). Along each elementary stream the pressure and temperature drop, and the gas velocity increases.

In flight due to deceleration of flow the streamlines part, forming a diffuser channel with "fluid" walls. In this case along each elementary stream the gas pressure and temperature rise, and velocity is lowered.

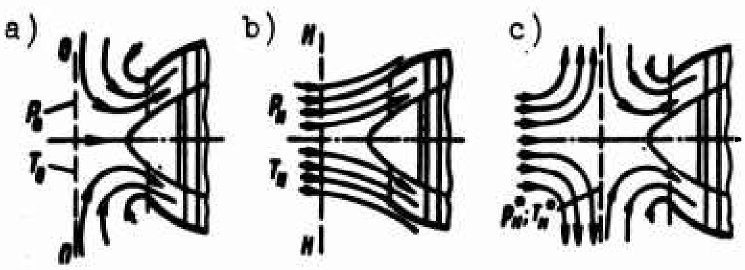


Fig. 2.20. Spectrum of streamlines at TRD inlet: a) during operation on a stand; b) in flight; c) model of general case.

Comparison of the streamline spectra when $c_0 = 0$ and $c_0>0$ shows that geometric and kinematic similarity of flows, entering the engine, do not exist, consequently, there is no gas-dynamic similarity. At supersonic flight speed shock waves appear at the engine inlet, which continuously (as the speed rises) introduce qualitative changes into the physical picture of streamline and change the fields of velocities, pressures and temperatures. From this it follows that subsonic and supersonic intake operating conditions in principle cannot be similar in the gas-dynamic relationship.

This can be generalized in the following way. Inasmuch as in undisturbed sections of flow on the stand and in flight the M numbers are not equal ($M_{c_0>0} \neq M_{c_0=0}$), gas-dynamic similarity of operating conditions of TRD inlet is impossible. At equal M_0 numbers the similarity of engine intake conditions is assured automatically. Kinematics of flow entering the TRD during flight can be conditionally reduced to operation of the intake on a stand, if we consider that a certain zone (region) always exists in front of the engine, on the boundary of which the flow is completely retarded ($c_0=0$, $p_H=p_H^*$, $T_H=T_H^*$). With respect to this zone the streamlines have precisely the same form as in the case of the flow of air on the stand. Thus, for flight at speed c_0 the process at the engine inlet can be represented as consisting of adiabatic deceleration (to parameters $c_0=0$, $T_H^*=T_H+\frac{c_0^2}{2g\frac{c_p}{A}}$, etc.) and subsequent acceleration of flow before the compressor inlet (Fig. 2.20c). Such a model of flow, being conditional, in a number of cases essentially simplifies analysis and calculations of TRD characteristics.

Jet Nozzle.

The flows of gas in TRD jet nozzle at various flight speeds are generally not similar, since on a stand and in flight the expansion ratios of gas in the jet nozzle are different,

$$\pi_{p.c}=\frac{p_5^*}{p_H}\neq \text{const}$$

and, consequently, M_5 numbers at the engine exit are different.

If, however, the convergent jet nozzle operates at critical pressure drops, i.e.,

$$\frac{p_5^*}{p_H}>\pi_{\kappa p}=\left(\frac{k+1}{2}\right)^{\frac{k}{k-1}},$$

then number $M_5 = 1 = \text{const}$ at the nozzle section and such a nozzle works under similar conditions.

For a Laval type jet nozzle similar conditions are conditions of underexpansion when $p_5 > p_H$.

Combustion Chamber.

Flows of gas in combustion chambers (or in afterburners) cannot be similar. Really, under similar conditions the following must be observed:

$$1)\ \frac{T_3^*}{T_2^*} = \text{const} \qquad 2)\ \alpha = \text{const}$$

(similarity of temperature fields) (constancy of excess air ratio, as a dimensionless parameter)

Bearing in mind that

$$\alpha \sim \frac{1}{(T_3^* - T_2^*)} = \text{const},$$

it is possible to arrive at the conclusion that observance of conditions 1) and 2) is possible only when $T_2^* = \text{const}$ and $T_3^* = \text{const}$.

However, this will no longer signify similarity, but the identity of gas flows.

It is easy to conclude that with change of engine operating conditions (for example, number of revolutions), and also flight conditions (for example, speed and altitude of flight), the operating conditions of the combustion chamber are changed.

Compressor.

Conditions of observance of the similarity of compressor operating conditions are the equality of two M numbers with respect to axial and peripheral velocities or values, proportional to them, i.e.,

1) $M_{1a} \sim q(\lambda_{1a}) \sim \bar{G}_{в} = \text{const};$
2) $M_{u} \sim \bar{n} \sim \lambda_{u} = \text{const}.$

Under similar conditions

$$\pi_{к}^{*} = \text{const};\ \eta_{iк}^{*} = \text{const};\ \bar{L}_{ад_{к}}^{*} = \text{const};\ \bar{N}_{к} = \text{const}\text{, etc.}$$

Turbine.

Conditions of observance of similarity of turbine operating conditions are also the constancy of two M numbers with respect to axial and peripheral velocities:

1) $M_{1a} \sim q(\lambda_{1a}) \sim \bar{G}_{г} = \text{const};$
2) $M_{u} \sim \bar{n} \sim \lambda_{u} = \text{const}.$

Under similar conditions

$$\pi_{т}^{*} = \text{const};\ \eta_{iт}^{*} = \text{const};\ \bar{N}_{т} = \text{const};\ \bar{L}_{ад_{т}}^{*} = \text{const}.$$

Turbojet Engine.

Conditions of observance of partial similarity of conditions of a geometrically fixed TRD are equalities:

1) $M_{0} = \text{const}$ and 2) $M_{u} = \text{const}.$

For the stand ($M_0 = 0$) there is required fulfillment only of condition $M_u = \text{const}$.

Under similar conditions the relative and dimensionless TRD parameters keep constant value.

Consequently, $\eta_0 = \text{const}$, $\bar{R} = \text{const}$, $\bar{C}_{уд} = \text{const}$, $\bar{G}_{т} = \text{const}$, etc.

Similarity Formulas of TRD

Using basic positions of the similitude theory, relationships can be found between TRD parameters under similar conditions. For this purpose it is necessary to represent the investigated engine parameter as a function of three flow parameters in any i section: pressure p_i, temperature T_i and velocity c_i. Then, using the property of similar flows – constancy of pressure and temperature ratios in any two sections and invariably of M_i numbers – express the TRD parameter as a function of values p_H^* and T_H^*.

The connection between gas parameters in sections i and H has the form:

$$\left.\begin{aligned} p_i &\sim p_H^* \\ T_i &\sim T_H^* \; . \\ c_i \sim \sqrt{T_i} &\sim \sqrt{T_H^*}, \text{ since } M_i=\text{const.} \end{aligned}\right\} \tag{2.26}$$

Similarity Formula for Thrust.

We have

$$R=\frac{G}{g}(c_3-c_0).$$

Expression

$$c_5-c_0\sim T_H^*,$$

and

$$G\sim\frac{p_3^*}{\sqrt{T_3^*}}\sim\frac{p_H^*}{\sqrt{T_H^*}}.$$

Then

$$R\sim p_H^*.$$

Consequently, under similar conditions

$$\frac{R}{p_H^*}=\text{const.} \tag{2.27}$$

Similarity Formula for Specific Fuel Consumption.

We have

$$C_{y\partial}=\frac{8.43c_0}{H_u\eta_0}\sim\sqrt{T_H^*}, \text{ since } \eta_i=\text{const and } H_u=\text{const.}$$

Consequently,

$$\frac{C_{y\partial}}{\sqrt{T_H^*}}=\text{const.} \tag{2.28}$$

Similarity Formula for Fuel Consumption per Hour (Second).

We have

$$G_T=RC_{y\partial}.$$

Using expressions (2.27) and (2.28), let us write

$$G_T\sim p_H^*\sqrt{T_H^*}.$$

Consequently,

$$\frac{G_T}{p_H^*\sqrt{T_H^*}}=\text{const.} \tag{2.29}$$

Similarity Formula for Specific Thrust.

We have

$$R_{y\partial}=\frac{c_5-c_0}{g}\sim\sqrt{T_H^*}.$$

Consequently,

$$\frac{R_{уд}}{\sqrt{T_H^*}} = \text{const.} \qquad (2.30)$$

Reduction of TRD Parameters to Standard Atmospheric Conditions

So that it would be possible to use TRD characteristics taken on a stand or in flight regardless of ambient conditions and that they would be universal, it is necessary to reduce results of tests, gas and engine parameters to standard atmospheric conditions.

Using similarity formulas for two conditions (measured and standard), formulas of reduction of TRD can be obtained.

Thrust Formula.

We have

$$\frac{R_{изм}}{p_H^*} = \frac{R_{пр}}{p_{ст}},$$

whence

$$R_{пр} = R_{изм}\frac{p_{ст}}{p_H^*} = R_{изм}\frac{760}{p_H^*}. \qquad (2.31)$$

For stand conditions ($c_0 = 0$).

$$R_{пр} = R_{изм}\frac{760}{p_н}. \qquad (2.32)$$

Formula of Specific Fuel Consumption.

We have

$$\frac{C_{уд_{изм}}}{\sqrt{T_H^*}} = \frac{C_{уд_{пр}}}{\sqrt{T_{ст}}},$$

whence

$$C_{уд_{пр}} = C_{уд_{зам}} \sqrt{\frac{T_{ст}}{T_H^*}} = C_{уд_{зам}} \sqrt{\frac{288}{T_H^*}}. \tag{2.33}$$

For stand conditions ($c_0 = 0$)

$$C_{уд_{пр}} = C_{уд_{зам}} \sqrt{\frac{288}{T_0}}. \tag{2.34}$$

Formula of Number of Revolutions.

We have

$$n_{пр} = n_{зам} \sqrt{\frac{288}{T_H^*}}. \tag{2.35}$$

For stand conditions

$$n_{пр} = n_{зам} \sqrt{\frac{288}{T_0}}. \tag{2.36}$$

Results of stand tests of TRD reduce to standard atmospheric conditions in the following way.

1. In accordance with the listed number of revolutions and atmospheric conditions we determine the physical number of revolutions that the engine should develop:

$$n_{зам} = n_{пр} \sqrt{\frac{T_0}{288}}.$$

2. At the obtained number of revolutions we measure engine parameters: thrust, specific fuel consumption and others.

3. By the reduction formulas we determine the values of engine parameters and compare them with the listed characteristic.

The divergence between listed parameters of various engines of the same series (when $n_{пр}=\mathrm{idem}$) should not exceed 0.5-1%. In this case the engines correspond to specification requirements.

Representation of TRD Characteristics in Similarity Parameters

So that TRD characteristics would be universal, they are constructed in similarity parameters in the form of dependences

$$R_{пр}=f_1(n_{пр}),\ C_{уд_{пр}}=f_2(n_{пр}),\ T^*_{3_{пр}}=f_3(n_{пр}).$$

It is easy to see how convenient it is to use universal TRD characteristics. The entire variety of throttle characteristics of thrusts, constructed for different values of T_0 and p_0 (Fig. 2.21), is converted into one thrust curve, represented in similarity parameters.

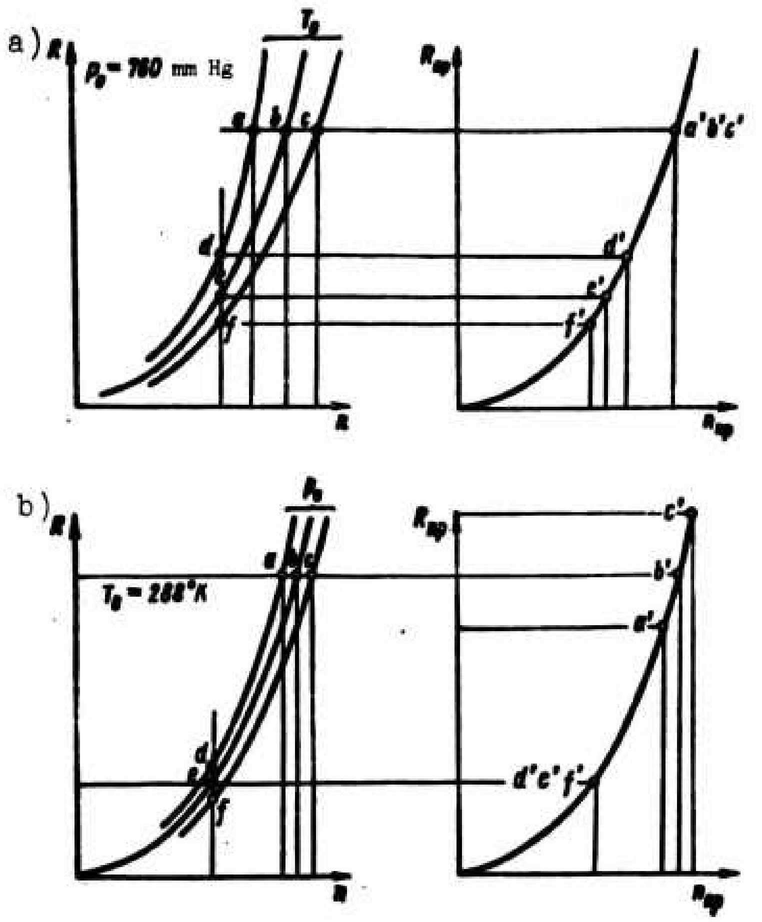


Fig. 2.21. Comparison of normal and universal TRD characteristics.

Let us dissect the family of characteristics $R = f(n, T_0)$ by a horizontal line (Fig. 2.21a) and make the obtained points of intersection by the letters a, b, c.

On universal characteristic $R_{\text{пр}}=f(n_{\text{пр}})$ these three points will be depicted in the form of one point a' (different values of physical revolutions n correspond to the same value of listed revolutions $n_{\text{пр}}$ at different temperatures T_0).

If we dissect the family of these characteristics by a vertical, then the obtained points of intersection d, e, f will be depicted on the universal TRD characteristic in the form of three points: d', e', f' with different values of listed revolutions (different values of $n_{\text{пр}}$ correspond to the same value of n at different temperatures T_0).

Let us now dissect the family of characteristics $R = f(n, p_0)$ by a horizontal (Fig. 2.21b) with intersecting points a, b, c. On the universal characteristic these points will also be depicted in the form of three points a', b', c' (different values of $n_{\text{пр}}$ correspond to different values of n when T_0 = const).

Points d, e, f, lying on the vertical line, will be depicted in the form of one point d' on the universal characteristic (since when T_0 = const and n = const we will also have $n_{\text{пр}}=\text{const}$).

CHAPTER III

HIGH-SPEED AND ALTITUDE CHARACTERISTICS OF TURBOJET ENGINES

§ 1. High-Speed Characteristics of Turbojet Engines

The dependences of thrust and specific fuel consumption on the flight speed with a prescribed program of engine control are called high-speed characteristics, or flight speed characteristics. High-speed characteristics are frequently supplemented by curves of change of gas temperature before the turbine, fuel consumption per hour, and also other important values in operation.

Programs of Flight Speed Control of TRD

There is a great variety of programs of turbojet engine flight speed control. These include control programs: for maximum thrust, for the best economy (minimum specific fuel consumption), constant geometry of the engine, preservation of total similarity of turbo-compressor operating conditions and different combined and special programs.

Let us consider the peculiarities of some of these programs, and also methods of their realization.

Program of TRD Control for Maximum Thrust.

Program of control of TRD for $R_{\text{макс}}$ assures automatically obtaining maximum thrust at all flight speeds and altitudes. For

its fulfillment it is necessary to observe the following conditions:

1) Maintaining maximum and constant engine revolutions

$$n=n_{макс}=\text{const}.$$

2) Maintaining maximum and constant temperature before the turbine

$$T_3^*=T_{3(макс)}^*=\text{const}.$$

Fulfillment of the first condition assures obtaining maximum airflow rate $G_в$ and maximum compression ratio, and the second (for example, with the aid of jet nozzle regulator) – jointly assures obtaining maximum specific thrust together with provision of $\pi_к^*=\pi_{к(макс)}^*$.

Thus, product

$$R_{уд_{макс}}G_{макс}=R_{макс}$$

turns out to be maximum at any prescribed values of H and c_0.

Maximum revolutions are maintained with the aid, for example, of a centrifugal speed regulator, interlinked with automatic fuel feed.

Maintaining maximum gas temperature T_3^* is a more complicated problem. It can be provided directly and indirectly.

For direct control of $T_3^*=\text{const}$ it is necessary to transmit a pulse, received from thermocouples installed at the nozzle box inlet of the turbine, to the automatic fuel feed for limitation or increase of the supply of combustible to the combustion chamber. Such automatic devices are installed on a number of domestic and foreign engines.

In a number of cases $T_3^*=\text{const}$ is sufficiently exactly preserved automatically. Let us assume that at all flight speeds the pressure drop in the jet nozzle is supercritical, i.e., $q(\lambda_5)=1$. Then on

the basis of equation (2.13) with increase of flight speed the pressure drop in the turbine $\pi_т^*$ also remains constant.

Consequently, from equality

$$L_т = L_к \text{ or } T_3^* = \frac{L_к}{118 \varepsilon_т^* \eta_т^*}$$

we find $T_3^* \sim L_к$.

If the compression ratio of the compressor is such that at n = const the compressor work at $n_{пр} = \text{var}$ remains constant ($\pi_к^* \approx 6$), i.e., $L_к = \text{const}$, then $T_3^* = \text{const}$.

Program of Control of TRD for the Best Economy.

By the best engine economy we mean the minimum specific fuel consumption at a prescribed flight speed.

If the TRD does not have special controlling elements,[1] $C_{уд} = C_{уд_{мин}}$ can be provided for each flight speed only by throttling the engine to revolutions corresponding to the best economy. For calculation of these conditions it is necessary to have the throttle characteristics of turbojet engines, constructed at various speeds and altitudes of flight; with their help we can obtain dependences $n = f_1(c_0)$, $C_{уд_{мин}} = f_2(c_0)$, and also $R = f_3(c_0)$.

Usually the TRD throttle characteristics at various flight speeds and altitudes are depicted in the form of dependences of specific fuel consumption on the degree of throttling with respect to thrust

$$C_{уд} = f(\bar{R}),$$

[1]Besides an automatic fuel meter, interlinked with the speed regulator.

where

$$\bar{R}=\frac{R}{R_{макс}}.$$

If, however, the TRD is equipped with a variable-area exhaust nozzle, then condition $C_{уд}=C_{уд_{мин}}$ can be provided by simultaneous regulation of the number of revolutions and the jet nozzle throat.

Program of TRD Control at Constant Engine Geometry.

Many of the contemporary turbojet engines do not have special control elements for the jet nozzle, compressor stator, etc. For these engines the change of basic flight speed and altitude parameters (R, $C_{уд}$, T_3^*, etc.) occurs automatically in accordance with limitations: n = const, f_5 = const, $\varphi_{НА}$ = const.

High-Speed Characteristics of Single-Shaft Nonreheated Turbojet Engines

Let us examine the high-speed characteristics of single-shaft nonreheated turbojet engines with control program for maximum thrust.

First let us sort out the peculiarities of TRD characteristics obtained by calculation without using characteristics of engine elements: compressor, combustion chamber, turbine and jet nozzle — considering only the change of gas-dynamic losses in the intake at supersonic flight speeds.

As basic conditions, forming the basis of calculation of these characteristics, let us take the following:

1) constant flight altitude H = const;

$$\left.\begin{array}{l} 2)\ n=n_{макс}=\text{const};\\[1ex] 3)\ T_3^*=T^*_{3_{макс}}=\text{const}. \end{array}\right\}\ \text{Control program for maximum thrust.}$$

Basic assumptions, usually applicable in approximation calculations of high-speed characteristics, include:

1) constancy of compressor work, i.e.,

$$L_{к}=\text{const} \text{ when } n=\text{const};$$

2) constancy of efficiency and loss factors of TRD elements:
$\eta_{к}^{*}=\text{const}$, $\eta_{т}^{*}=\text{const}$, $\sigma_{к.с}^{*}=\text{const}$, $\varphi_{р.с}=\text{const}$, $\xi_{к.с}=\text{const}$.

The assumption about constancy of efficiency of the compressor is the roughest. The effect of $\eta_{к}^{*}=\text{var}$ on TRD high-speed characteristic is examined below;

3) total expansion of gas in the jet nozzle of TRD:

$$p_5=p_H.$$

The last assumption presupposes a device such as an all-condition Laval type variable-area exhaust nozzle.

The Effect of Flight Speed on the Compression Ratio of Air in TRD

With increase of flight speed the dynamic compression ratio continuously increases:

$$\pi_{д}=\frac{p_1^{*}}{p_H}=(1+0{,}2M_H^2)^{3,5}\sigma_{д}^{*},$$

as does the kinetic heating of air in front of the engine.[1]

Increase of stagnation temperature at compressor inlet

$$T_H^{*}=T_H(1+0{,}2M_0^2)$$

[1]Dependence of $\sigma_{д}^{*}$ on the Mach number of flight for different systems of shock waves in the TRD intake is given, for example, in [2].

at the same amount of expended work

$$L_к = 102{,}5T^*_H(\pi_к^{*0.286} - 1)\frac{1}{\eta^*_к} = \text{const}$$

leads to the fact that the compression ratio of the compressor drops monotonously; with increase of M_0 the value of $\pi^*_к$ asymptotically approaches one.

However, the total compression ratio of air

$$\pi = \pi_д \pi^*_к$$

rises in this case, since increase of $\pi_д$, is determining, which turns out to be even greater, the less the losses in the appearing shock waves of the intake.

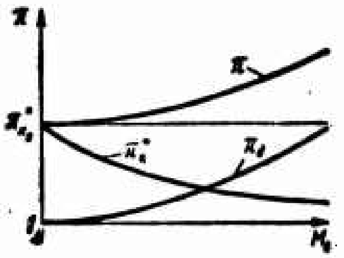


Fig. 3.1. Effect of flight speed on compression ratio of TRD.

Figure 3.1 contains curves of change of compression ratios π, $\pi^*_к$ and $\pi_д$ as a function of flight speed.

Change of Specific Thrust

With increase of flight speed the total compression ratio, and consequently also the total expansion ratio are continuously increased. Since the pressure drop on the turbine remains constant (when $\lambda_5 \geqslant 1$ always $\pi^*_т = \text{const}$), the expansion ratio of gas in the jet nozzle grows in proportion to the compression ratio:

$$\pi_{р.с} = \frac{p^*_4}{p_H} = \frac{\pi_p}{\pi^*_т} = \frac{\pi \sigma^*_{к.с}}{\pi^*_т} \sim \pi.$$

In this case the gas temperature past the turbine keeps a constant value

$$T_4^* = T_3^* - \frac{L_T}{118} = \text{const} \quad (\text{since } T_3^* = \text{const and } L_T = \text{const}).$$

As a result the exit velocity of gas from the jet nozzle, being

$$c_5 = \varphi \sqrt{2g\frac{c_p}{A}T_4^*\left[1 - \left(\frac{p_H}{p_4^*}\right)^{\frac{k-1}{k}}\right]},$$

is also continuously increased.

Specific thrust of TRD

$$R_{y\partial} = \frac{c_5 - c_0}{g}$$

is lowered with rise of flight speed, since increase of the exit velocity from the nozzle occurs considerably slower than the increase of flight speed.

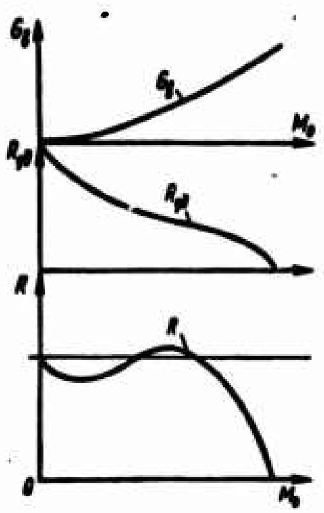


Fig. 3.2. Effect of flight speed on the specific thrust, air-flow rate and total thrust of TRD.

Decrease of specific thrust of TRD with rise of flight speed (Fig. 3.2) can be explained in the following way. With increase of the compression ratio when $T_3^* = \text{const}$ the effective work of the cycle (being close to maximum on a stand) drops, approaching

zero. Consequently, the specific thrust, as ensues from expression

$$R_{уд}=\frac{1}{g}\left(\sqrt{2gL_e+c_0^2}-c_0\right),$$

also approaches zero with increase of c_0.

Change of Airflow Rate

With increase of flight speed the mass flow rate of air through the engine continuously increases in proportion to the total compression ratio. This may be seen from the equation of flow rate of gas for the turbine nozzle box

$$G_в\approx G_г=m\frac{p^*_{CA}}{\sqrt{T^*_{CA}}}f_{CA}q(\lambda_{CA}),$$

whence we obtain that

$$G_в\sim p^*_2\sim\pi,$$

since $T^*_{CA}=T^*_3=\mathrm{const},\ q(\lambda_{CA})=1=\mathrm{const},\ p_H=\mathrm{const}.$

Change of the airflow rate as a function of flight speed is shown on Fig. 3.2.

Change of Total Thrust

Change of total thrust of TRD as a function of flight speed is determined by regularities of change of its cofactors — specific thrust and airflow rate:

$$R=R_{уд}G_в.$$

With increase of c_0 in the region of low M_0 (<0.4-0.5) numbers the thrust at first drops, since lowering of specific thrust is still not compensated by increase of the mass flow rate of air; in the transonic and supersonic regions of flight the TRD thrust is increased and at small compression ratios of the compressor ($\pi^*_{к_0}=3$—5) can considerably exceed the bench value R_0. Such

regularity is explained by the intense rise of the airflow rate combined with a more moderate drop of specific thrust. Finally, at high supersonic flight speeds the TRD thrust, having reached maximum, drops right to zero in accordance with the inevitable tendency of specific thrust change.

Change of TRD Efficiency

Let us examine how TRD efficiencies – effective, thrust and overall (Fig. 3.3) – are changed depending on the flight speed.

Effective Efficiency.

From expression

$$\eta_e = \frac{AL_e'}{q_{вн}} = \frac{A\,\dfrac{c_5^2 - c_0^2}{2g}}{\dfrac{c_{pm}(T_3^* - T_2^*)}{\zeta_{к.с}}}$$

it follows that the change of effective efficiency is determined by the regularity of change of effective work of the cycle and introduced heat (with fuel) with respect to the flight speed.

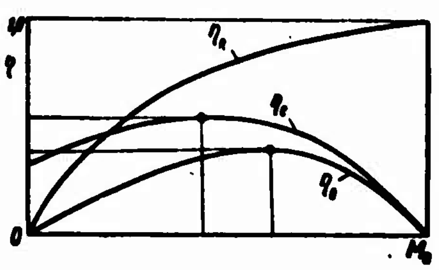


Fig. 3.3. Change of TRD efficiency with respect to flight speed.

Since the amount of heat introduced to each kilogram of air is continuously decreased with rise of c_0 (because of increase of the air temperature at the burner inlet), and the work of the cycle hardly changes at subsonic flight speeds, then the value of η_e is increased at first. At supersonic flight speeds the increasing drop of cycle work leads to continuous lowering of parameter η_e (right to zero).

Thrust Efficiency.

The value of thrust efficiency is determined from expression

$$\eta_R = \frac{L_R}{L_e} = \frac{2c_0}{c_0 + c_5}.$$

When $c_0 = 0$ we have $\eta_R = 0$, when $c_0 = c_5$ we obtain $\eta_R = 1$.

Consequently, with increase of flight speed the value of thrust efficiency increases from zero to 1 (see Fig. 3.3). Thus, losses of energy with discharge velocity are continuously decreased as the flight speed rises. At the moment when they completely vanish and the thrust efficiency reaches the highest possible theoretical value, engine thrust vanishes.

Overall Efficiency.

From expression

$$\eta_0 = \frac{AL_R}{q_{вн}} = \eta_e \eta_R$$

it follows that change of overall efficiency of TRD as a function of flight speed is determined by change of the internal and external efficiency (see Fig. 3.3).

When $c_0 = 0$ and $c_0 = c_5$ the overall efficiency becomes zero. With increase of c_0 the overall efficiency rises, reaches maximum at high supersonic speed, and then drops to zero.

Change of the overall efficiency characterizes the TRD economy in the entire flight speed range.

At number $M_0 = 2.2$ the value of $\eta_{0(\text{макс})}$ of the best turbojet engines reaches 0.40-0.45.

Change of Specific Fuel Consumption

Let us find the connection between specific fuel consumption and flight speed.

We have

$$C_{\text{уд}}=3600\,\frac{m_{\text{т}}}{R_{\text{уд}}}.$$

Having replaced $m=\frac{q_{\text{нн}}}{H_{\text{н}}}$ and having multiplied the numerator and denominator by Ac_0, where $A=\frac{1}{427}$ kcal/kgf, we obtain

$$C_{\text{уд}}=8{,}43\,\frac{c_0}{H_{\text{н}}\eta_0}. \qquad (3.2)$$

When $c_0 = 0$ value $\eta_0 = 0$. In this case expression (3.2) is turned into indeterminacy; for expanding it in equation (3.2) let us place expression

$$\eta_0=\eta_e\,\frac{2c_0}{c_5+c_0}.$$

Then we obtain

$$C_{\text{уд}_0}=8{,}43\,\frac{c_0(c_5+c_0)}{\eta_e 2c_0}=4{,}215\,\frac{c_{5_0}}{\eta_{e_0}}. \qquad (3.3)$$

Thus, with increase of flight speed from 0 to c_5 the specific fuel consumption rises continuously from the initial bench value, becoming infinity when $c_0 = c_5$ (Fig. 3.4). Increase of specific

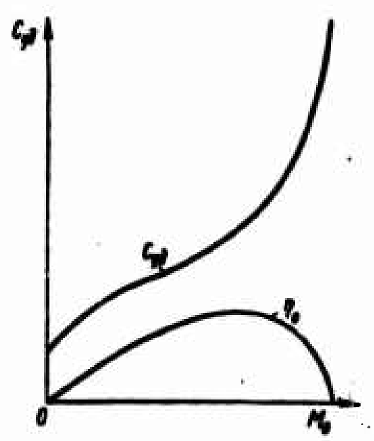


Fig. 3.4. Change of specific fuel consumption with respect to flight speed.

fuel consumption depending on the flight speed does not attest to continuous impairment of the economy of TRD operation, since parameter $C_{уд}$ is not a criterion of economy. Economy of TRD operation deteriorates only in the flight speed range at which a drop of overall efficiency sets in. In the flight speed range from zero to "economic" speed (at which $\eta_0=\eta_{0\,макс}$) the TRD economy continuously increases.

What then is the physical meaning of continuous increase of $C_{уд}$? It involves the fact that the work of every kilogram of thrust is continuously increased with rise of flight speed $L_{(R=1)}=1\cdot c_0$.

From this it follows that the expended thermal energy, proportional to fuel consumption for obtaining this kilogram of thrust, should be increased and in view of this the specific fuel consumption $C_{уд}$ should grow.

The Effect of Parameters of the Process (Gas Temperature Before the Turbine and Bench Compression Ratio of Compressor) on the Peculiarity of High-Speed Characteristics of TRD

The cited dependences of basic TRD flight speed parameters carry a fundamental character. However, the numerical values of specific and dimensionless parameters ($R_{уд}$, $C_{уд}$, η_e, η_R, η_0), characterizing the perfection of the TRD, values of flight speeds at which TRD thrust becomes zero, depend on the level of parameters of the TRD working process.

The Effect of $\pi^*_{к_0}$.

To every value of complex $\frac{T^*_3\eta_p\eta_c}{T_H}$ there corresponds a fully determinate limiting value of the total compression ratio of TRD equal to

$$\pi_{макс}=\left(\frac{T^*_3\eta_p\eta_c}{T_H}\right)^{3,5}=\pi^*_к\pi_д=\text{const},$$

at which $c_5 = c_0$ and the effective work of the cycle becomes zero ($L_e = 0$). Consequently, with increase of the compression ratio of compressor a smaller dynamic compression ratio is required, i.e., lower flight speed $c_{0\,\text{макс}}$ for achievement of $\pi_{\text{макс}}$.

Thus, with increase of $\pi^*_{к_0}$ the specific thrust, and thus total thrust, drop more sharply with increase of flight speed, reaching zero at low values of $c_{0\,\text{макс}}$ (Fig. 3.5). In accordance with the

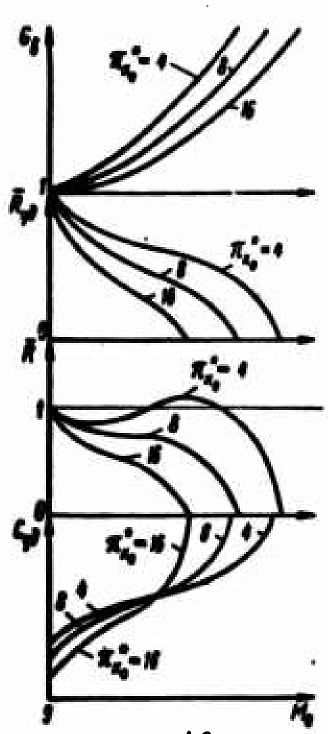


Fig. 3.5. The effect of calculated values of $\pi^*_{к_0}$ on high-speed characteristics of TRD.

sharper drop of $R_{уд}$ the value of specific fuel consumption increases faster with respect to flight speed, approaching infinity at low values of $c_{0\,\text{макс}}$. In this case the bench value $C_{уд}$ turns out to be smaller, the higher the value of $\pi^*_{к_0}$ is.

Curves of the flow rate of air also depend on the numerical value of $\pi^*_{к_0}$. The larger is $\pi^*_{к_0}$, the smaller the portion of dynamic compression as compared to mechanical, and the slower the airflow rate is increased with respect to flight speed.

Thus, sharper drop of $R_{уд}$ and delayed rise of $G_в$ correspond to large values of $\pi^*_{к_0}$. This circumstance causes deformation of

TRD thrust curves — with increase of $\pi^*_{\kappa_0}$ the "hump" on the high-speed characteristics gradually vanishes. Already at $\pi^*_{\kappa_0} > 10$—12 and usual values of T^*_3 ($\leqslant 1200°$ K) the thrust drops continuously in the whole flight speed range (see Fig. 3.5).

Analysis of Fig. 3.5 shows that the application of high values of $\pi^*_{\kappa_0}$ (>12—15) sharply improves TRD economy on the stand and at subsonic flight speeds. The transition to lowered values of $\pi^*_{\kappa_0}$ (4—8) gives the possibility of improving thrust characteristics, and increasing the economy of engine operation at supersonic flight speeds.

The Effect of T^*_3 (Fig. 3.6).

With increase of the gas temperature before the turbine the effective work of the cycle and the exit velocity of gas from the

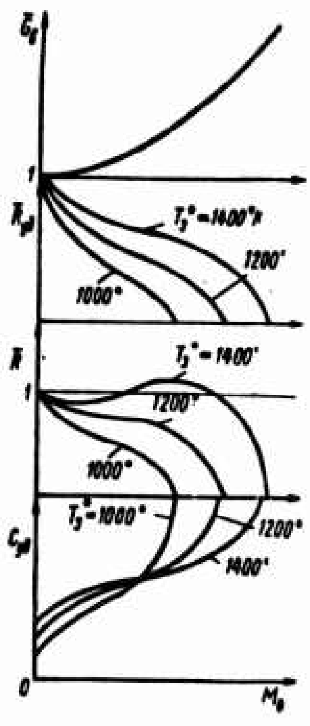


Fig. 3.6. Effect of calculated values of T^*_3 on high-speed characteristics of TRD.

jet nozzle increase. In this case the drop of specific thrust with respect to flight speed is slowed. As a result the specific and total thrust of TRD become zero at high values of $\lambda_{макс}$ and $c_{0_{макс}}$.

Accordingly, specific fuel consumption reaches infinity also at high values of c_0. With increase of T_3 the bench value of $C_{y.d}$ continuously increases.

It is characteristic that increase of T_3^* does not affect the regularity of change of the airflow rate with respect to the flight speed. However, the characteristic of total thrust becomes more favorable.

From the above the following conclusion can be made: for improvement of economy of TRD operation on a stand and at subsonic flight speeds one should apply high values of π_K^* and low values of T_3^* (1000-1200°K). Application of small values of π_K^* and high of T_3^* (>1250–1300° K) gives the possibility of essentially improving the economy of TRD and the shape of thrust characteristics of the engine at high supersonic M_0 numbers.

Peculiarities of High-Speed Characteristics of Two-Shaft Turbojet Engines

Let us examine the peculiarities of high-speed characteristics of two-shaft TRD with two programs of control:

1) $n_{\text{ВД}}=\text{const}$ and $f_5=\text{const}$;

2) $n_{\text{НД}}=\text{const}$ and $f_5=\text{const}$.

First of all let us clarify how the number of revolutions of the free compressor stage and gas temperature before the turbine are changed with respect to the flight speed.

A diagram of a two-shaft TRD with designations of characteristic sections of the gas-air duct is shown on Fig. 1.2b. We will consider that at all flight speeds and altitudes the jet nozzle operates at supercritical pressure drop $[q(\lambda_5)=1]$.

Program $n_{ВД} = \text{const}$ and $f_5 = \text{const}$

With increase of flight speed there sets in "loading" of the low-pressure compressor stage (blade angles of attack increase and as a result the compression ratio and work of the low-pressure compressor are increased) and "lightening" of high-pressure compressor stage (blade angles of attack are decreased and as a result the values of $\pi^{*}_{КВД}$ and $L_{КВД}$ are lowered). In accordance with this, the lines of operating conditions of the low-pressure compressor (KND) and the high-pressure compressure (KVD) are deformed, being deflected from the direction they would occupy if the TRD consisted of only a low-pressure or high-pressure turbocompressor (the joint influence of compressor stages would be excluded).

Lowering the work of a high-pressure compressor when $n_{ВД} = \text{const}$ causes decrease of fuel feed to the combustion chamber and lowering of T^{*}_{3}. Really, from the work balance of high-pressure compressor

$$L_{КВД} = L_{ТВД} = 118 T^{*}_{3} \varepsilon_{ТВД} \eta_{ТВД}$$

when $\pi^{*}_{ТВД} = \text{const}$ it follows that

$$L_{КВД} \sim T^{*}_{3}.$$

Decrease of T^{*}_{3} in turn leads to some lowering of the gas temperature before the low-pressure turbine $T^{*}_{4ВД}$. Consequently, the low-pressure turbine is "lightened" — its work is decreased. Since the low-pressure compressor is "loaded" with increase of c_0, the advanced unbalance of works

$$L_{КНД} > L_{ТНД}$$

is removed only by lowering the revolutions of the low-pressure stage.

As a result, with increase of flight speed the revolutions of the low-pressure compressor and the gas temperature before turbine drop. Performance points of the low- and high-pressure compressors are shifted along the lines of operating conditions toward lowered listed revolutions.

Program $n_{\text{НД}} = \text{const}$ and $f_5 = \text{const}$

With increase of flight speed, as in the preceding case, there sets in "loading" of the low-pressure compressor stage and "lightening" of the high-pressure compressor stage.

Increase of the work of the low-pressure compressor when $n_{\text{НД}} = \text{const}$ causes additional fuel feed to the combustion chamber and, consequently, increase of T_3^* and $T_{4\text{ВД}}^*$. Really, from the work balance of the low-pressure compressor

$$L_{\text{КНД}} = L_{\text{ТНД}} = 118 T_{4\text{ВД}}^* \varepsilon_{\text{ТНД}} \eta_{\text{ТНД}}^*,$$

when $\pi_{\text{ТНД}}^* = \text{const}$ it follows that

$$L_{\text{КНД}} \sim T_{4\text{ВД}}^* \sim T_3^*.$$

Lightening of the high-pressure compressor with simultaneous increase of $L_{\text{ТНД}}$ (because of rise of T_3^*) leads to an unbalance of works on the high-pressure turbocompressor, i.e.,

$$L_{\text{КВД}} < L_{\text{ТВД}}.$$

This is overcome by acceleration of the high-pressure rotor.

As a result with increase of flight speed the revolutions of the high-pressure compressor increase and the gas temperature before the turbine rises.

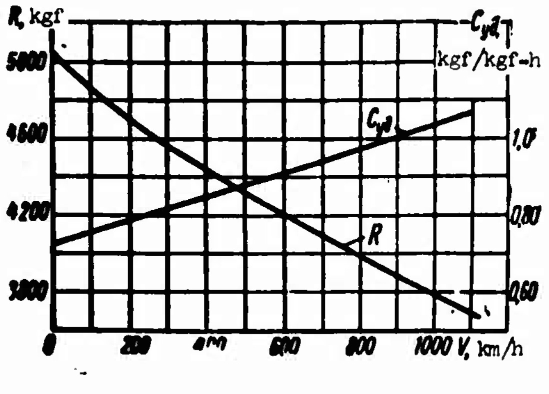


Fig. 3.7. High-speed characteristic of "Avon" TRD.

As an illustration of high-speed characteristic of a single-shaft TRD on Fig. 3.7 there is shown the characteristic of the English engine "Avon."

§ 2. Altitude Characteristics of Turbojet Engines

Dependences of thrust and specific fuel consumption on the altitude of flight at a prescribed program of engine control are called altitude characteristics, or flight altitude characteristics of turbojet engines. Altitude characteristics, similar to high-speed, are often supplemented by curves of change of the gas temperature before the turbine, consumption of fuel per hour, and also other parameters that are important in operation.

During climb the same program of engine control is used as for the flight speed. We will limit ourselves here to analysis of the altitude characteristics of single-shaft TRD, constructed for the program of control of maximum thrust.

Altitude Characteristics of Single-Shaft Nonreheated Turbojet Engines

Let us examine the altitude characteristics of single-shaft nonreheated turbojet engines, constructed on the basis of approximate thermal and gas-dynamic calculation of the engine with a program of maximum thrust control.

As basic conditions for construction of altitude characteristics let us take:

constant flight speed c_0 = const;
constant number of revolutions $n = n_{\text{макс}} = \text{const}$;
constant gas temperature before the turbine $T_3^* = T_{3_{\text{макс}}}^* =$ const.

Assumptions that are usually taken in approximate calculations of altitude and high-speed characteristics, as before, will include:

constancy of compressor work at constant revolutions;
constancy of efficiency and loss factors of TRD elements;
total expansion of gas in the TRD jet nozzle.

The assumption of constancy of compressor and turbine efficiency at high altitudes ($H > 15$ km) is rather approximate. It is known that at high altitudes a drop of Reynold's numbers sets in, referred to blade chord — characteristic linear dimension of compressor and turbine grids. As a result of this, for small turbojet engines a drop of efficiency of the indicated elements[1] can set in.

The assumption about invariability of the combustion efficiency of the engine during climb also turns out to be approximate. At very high altitudes the lowering of pressure in the fuel system and in combustion chambers leads to sharp impairment of carburation and to a drop of mechanical combustion efficiency of the fuel. Maintaining high combustion efficiency requires the application of special vaporizing combustion chambers, multichannel injectors, catalysts of combustion reaction, etc.

The Effect of Flight Altitude on the Compression Ratio of the TRD

With climb to isothermal limit ($H = 11$ km) the ambient temperature T_H continuously decreases. Therefore, the stagnation temperature of air at the compressor inlet drops,

$$T_1^* = T_H^* = T_H + \frac{c_0^2}{2g \cdot \frac{c_p}{A}}.$$

The shown circumstance leads, on the one hand, to increase of M_0 number of flight[2] and to increase of dynamic compression ratio

$$\pi_д = (1 + 0{,}2M_0^2)^{3,5}\sigma_д^*,$$

[1]This question is discussed in Chapter VII in greater detail.

[2]Since the speed of sound is decreased.

where

$$M_{H} = \frac{c_0}{\sqrt{kgRT_H}},$$

and on the other, – to increase of the compression ratio of the compressor.

Indeed, from consideration of equality

$$L_{\kappa} = 102{,}57\, T^{*}_{H}\left(\pi_{\kappa}^{*0.286} - 1\right)\frac{1}{\eta^{*}_{\kappa}} = \text{const}$$

it is easy to conclude that with lowering of T^{*}_{H} the value of π^{*}_{κ} increases.

Thus, total compression ratio

$$\pi = \pi_{д}\pi^{*}_{\kappa}$$

is also increased (Fig. 3.8).

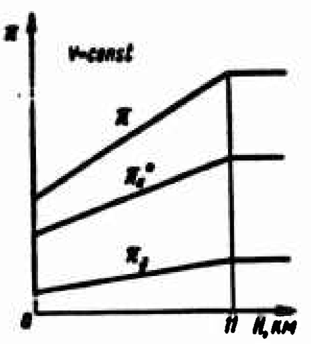


Fig. 3.8.

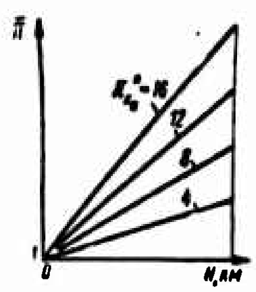


Fig. 3.9.

Fig. 3.8. The effect of flight altitude on the compression ratio of TRD.

Fig. 3.9. The effect of the computed value of $\pi^{*}_{\kappa_0}$ on the change of compression ratio during climb.

Increase of the total compression ratio turns out to be more considerable, the higher the absolute value of bench compression ratio of the compressor ($\pi^{*}_{\kappa_0}$) and the lower the flight speed (Fig. 3.9).

Change of Airflow Rate

Let us now examine how the mass flow rate of air through the TRD is changed with climb (Fig. 3.10).

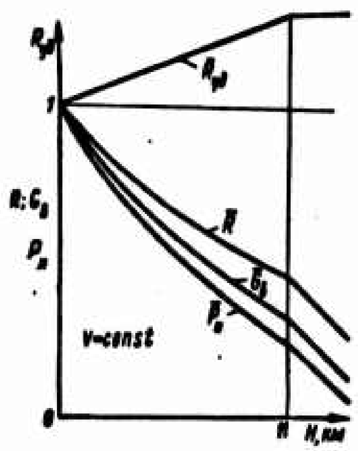


Fig. 3.10. The effect of flight altitude on the specific and total thrust of TRD and the flow rate of air through the engine.

From the expression of airflow rate, written for the turbine nozzle box

$$G_{\text{в}} = m \frac{p_3^* \sigma_{\text{CA}}}{\sqrt{T_3^*}} f_{\text{CA}} q(\lambda_{\text{CA}}),$$

it follows that

$$G_{\text{в}} \sim p_3^* \sim \pi p_H. \tag{3.4}$$

Thus, the flow rate of air through the engine is decreased during climb; however, it is slower than the external pressure drops. The factor, delaying the drop of $G_{\text{в}}$, turns out to be the increase of total compression ratio, which is caused by lowering of the ambient temperature of the atmosphere up to H = 11 km. At altitude H = 11 km the relative pressure is

$$\bar{p}_H = \frac{p_{H=0}}{p_{H=11\,\text{км}}} = 0{,}223.$$

For $\pi_{\text{к}_0}^* = 10$, number $M_0 = 0.9$, H = 11 km and $L_{\text{к}} = \text{const}$ we find relative compression ratio $\bar{\pi} = 1.57$.

Then $\overline{G}_{в} = \overline{p}_{H}\overline{\pi}$ = 0.223·1.57 = 0.35, i.e., with a fivefold drop of external pressure the flow rate of air drops 65%.

Change of Specific Thrust (See Fig. 3.10)

At constant flight speed the change of specific thrust depends only on the exit velocity of gas from the nozzle.

It is not difficult to conclude that at conditions $T_3^* = \text{const}$ and $L_к = \text{const}$ the exit velocity of gas depends only on the pressure drop in the jet nozzle, i.e.,

$$c_5 \sim \sqrt{e_{p.c}},$$

where

$$e_{p.c} = 1 - \frac{1}{(\pi_{p.c})^{\frac{k-1}{k}}};$$

$$\pi_{p.c} = \frac{\pi \sigma_{к.с}^*}{\pi_т^*} \sim \pi \text{ (assuming that } \pi_т^* = \text{const).}$$

Since the compression ratio increases during climb to the isothermal limit, the exit velocity from the engine, and consequently also the specific thrust are increased. Increase of specific thrust turns out to be more intense, the higher the ground compression ratio of the compressor ($\pi_{к_0}^*$), the lower the gas temperature before the turbine (T_3^*), and the higher the flight speed.

With climb to H = 11 km the specific thrust increases in the usual range of values of $\pi_{к_0}^*$ and T_3^* by 40-60%.

Change of Total Thrust

Change of total thrust (see Fig. 3.10) with climb is determined by change of its components

$$\overline{R} = \overline{R}_{уд}\overline{G}_{в} = f(H).$$

The basic factor of the change of total thrust with respect to flight altitude is the drop of airflow rate, caused by continuous decrease of external pressure. However, the drop of thrust occurs slower than lowering of $G_в$. It lags behind the rise of specific thrust. At a fivefold decrease of external pressure the airflow rate drops approximately three times, and TRD thrust not more than two times.

The drop of thrust is intensified with decrease of flight speed, lowering of $\pi^*_{к_0}$ and rise of T^*_3.

Change of TRD Thrust at Altitudes Higher Than 11 km

At altitudes higher than 11 km the external temperature keeps a constant value $T_H = 216.5^oK$ right up to altitude H = 25-30 km. At these altitudes (in the entire isothermal region) we have $\pi = \text{const.}$ Consequently, specific thrust is also constant here. Drop of the airflow rate and total thrust of the engine is accelerated. Now it occurs in proportion to the atmospheric pressure, i.e.,

$$\bar{G}_{в_{H>11\,км}} = \bar{R} = \bar{p}_H. \qquad (3.5)$$

Change of Efficiencies

<u>Effective Efficiency</u>.

During climb the exit velocity of gas from the nozzle increases, and consequently, the kinetic energy of each kilogram of gas at the engine exit is increased. On the other hand, the amount of heat introduced with fuel to this kilogram of gas in the combustion chamber increases (since the preheating interval grows $\Delta T^*_{к.с} = T^*_3 - T^*_2$). In view of the fact that L_e is increased faster than $q_{вн}$, the effective efficiency η_e of TRD increases during climb (Fig. 3.11).

Increase of η_e with respect to altitude can also be explained as the result of simultaneous increase of total compression

ratio (π) and the degree of preheating $\left(\Delta = \frac{T_3^*}{T_H}\right)$ of the working medium per cycle.

Thrust Efficiency.

Increase of exit velocity from the jet nozzle at constant flight speed leads to increase of losses with exit velocity and to drop of thrust efficiency

$$\eta_R = \frac{2c_0}{c_0 + c_5}$$

with climb (see Fig. 3.11). The closer η_R approaches one, the less it is affected by increase of c_5.

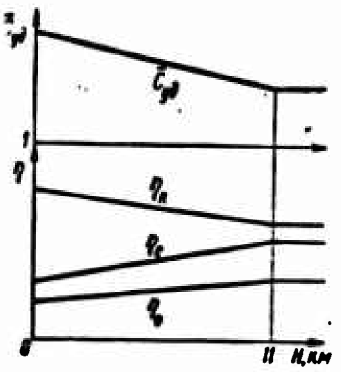


Fig. 3.11. The effect of flight altitude on the efficiency and specific fuel consumption of TRD.

Overall Efficiency.

Change of overall efficiency η_0 depends on the change of its cofactors

$$\eta_0 = \eta_e \eta_R.$$

Calculations show that at high values of η_R, increase of effective efficiency is the determining element. Thus, during climb the overall efficiency of TRD will rise. This fact characterizes the improvement of economy of engine operation at altitude.

Since at constant flight speed the specific fuel consumption is inversely proportional to overall efficiency

$$C_{уд}=8{,}43\frac{c_0}{H_u\eta_0}\sim\frac{1}{\eta_0},$$

during climb to H = 11 km the value of $C_{уд}$ is continuously lowered (see Fig. 3.11).

Further climb (when $H>11$ км) does not affect the value of $C_{уд}$. Cases when impairment of combustion efficiency sets in, as a result of which $C_{уд}$ increases, are an exception, since

$$C_{уд}\sim\frac{1}{\zeta_{к.с}}.$$

As an illustration of altitude characteristic of TRD on Fig. 3.12 there is shown the characteristic of "Avon" engine.

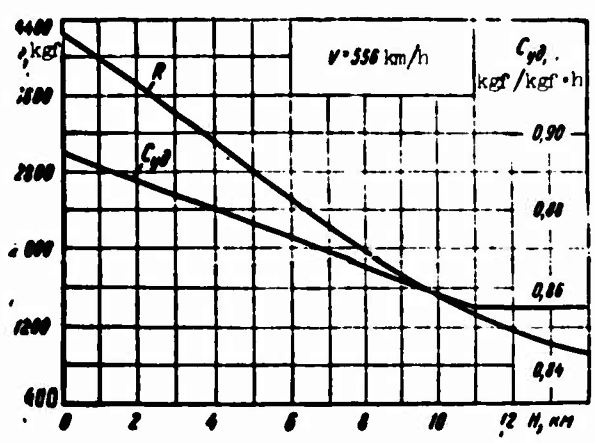


Fig. 3.12. Altitude characteristic of "Avon" turbojet engine.

§ 3. Peculiarities of High-Speed and Altitude Characteristics of TRDF

Program of Control of TRDF for Maximum Thrust

Program of control of TRDF for maximum thrust stipulates

simultaneous fulfillment of the following conditions:

$$n=n_{\text{макс}}=\text{const},\ T_3^*=T_{3(\text{макс})}^*=\text{const и } T_\Phi^*=T_{\Phi_{\text{макс}}}^*=\text{const}.$$

Fuel Flow Control to the Afterburner

In connection with the necessity of maintaining constant gas temperature at the afterburner exit with respect to speed and altitude of flight a question appears about the rational substantiation of the principle of control of parameters of the gas-air duct, assurring $T_\Phi^*=\text{const}$ at a fixed throat area of the jet nozzle. Such a principle is automatically maintaining constant pressure drop on the turbine, i.e., providing $\pi_T^*=\text{const}$ during fuel feed to the afterburner.

Really, preservation of $\pi_T^*=\text{const}$ when $n=\text{const}$ and $L_K=\text{const}$ provides $T_3^*=\text{const}$ and $T_4^*=\text{const}$. At a fixed position of the jet nozzle of the engine any change of fuel feed, disturbing condition $T_\Phi^*=\text{const}$, leads to thermal control of the engine, i.e., change of compressor and turbine conditions. For example, with increase of T_Φ^* the backpressure of gas past the turbine rises the same as occurs when the jet nozzle is covered. In this case the pressure drop on the turbine π_T^* is decreased, the performance point of the compressor is displaced toward the surge limit. It is obvious that preservation of $\pi_T^*=\text{const}$ can be the result of only the fuel feed at which $T_\Phi^*=\text{const}$ is ensured.

High-Speed Characteristic of TRDF

With increase of flight speed the specific thrust of TRDF

$$R_{y\text{д}_\Phi}=\frac{c_{5\Phi}-c_0}{g}$$

drops, however, due to the high exit velocity of gas its drop occurs considerably slower than for a turbojet engine. Regularity of change of the airflow rate in relation to the flight speed for TRDF and TRD coincides. As a result the total thrust of TRD is

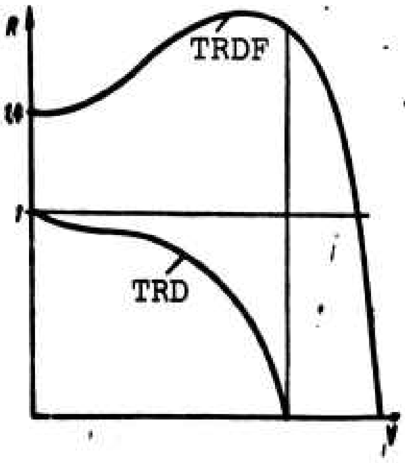


Fig. 3.13.

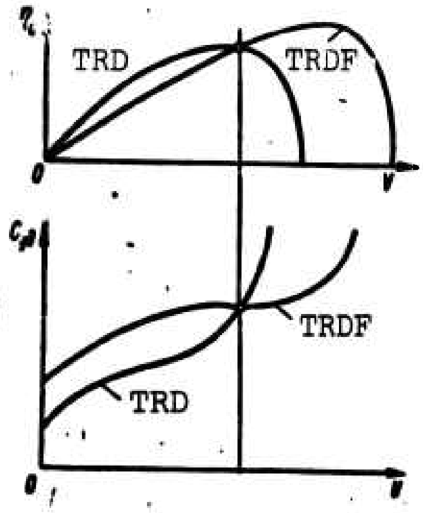


Fig. 3.14.

Fig. 3.13. High-speed characteristics of TRDF thrust with the afterburner turned on and turned off.

Fig. 3.14. Change of overall efficiency and specific fuel consumption of TRD and TRDF depending on the flight speed.

at first little changed with increase of speed, and then rises considerably; only in the region of high M_0 numbers does thrust sharply drop, approaching zero.

Figure 3.13 shows high-speed characteristics of TRDF with the afterburner switched on and turned off. It is clear that in the entire range of M_0 numbers of flight the thrust of the TRDF is considerably greater than for the TRD.

Specific fuel consumption of TRDF

$$C_{у д_{ф}} = 3600 \frac{m_{т ф}}{R_{у д_{ф}}} = 8,43 \frac{V}{H_u \eta_{п}}$$

increases with increase of flight speed.

However, at supersonic flight speeds a sharp increase of effective, and consequently overall efficiency can lead even to some "local" lowering of $C_{у д_{ф}}$.

Figure 3.14 shows change of overall efficiency and specific fuel consumption of TRDF with the afterburner switched on and off depending on the flight speed. If on the stand and at subsonic flight speeds the TRD is more economical than the TRDF, on high M_0 numbers of flight (>3) the specific fuel consumption of TRDF is lower than for the TRD. This is explained by the fact that increase of specific thrust of TRDF on these speeds compensates the additional fuel consumption with excess. Effective efficiency of the TRDF cycle is now even higher than for the TRD.

Peculiarities of Altitude Characteristics of TRDF

Altitude characteristics of TRDF with a control program for maximum thrust do not practically differ from TRD characteristics. During climb the TRDF thrust drops continuously; above 11 km the drop of thrust is accelerated; it adheres to regularity

$$R_{H>11\,\text{км}} \sim p_H.$$

Regarding the specific fuel consumption, during the climb it is lowered right up to H = 11 km. Further climb leads to deterioration of the economy of engine operation only in case of a decrease of combustion efficiency.

Degree of Thrust Augmentation

The degree of thrust augmentation of TRDF is the name for the ratio of augmented thrust to initial, nonaugmented, engine thrust

$$\bar{R}_{\text{ф}} = \frac{R_{\text{ф}}}{R}. \tag{3.6}$$

If with the afterburner switched on the operating conditions of the turbocompressor, and consequently, the airflow rate through the engine remain constant, then

$$\bar{R}_{\text{ф}} = \frac{R_{\text{уд ф}}}{R_{\text{уд}}} = \frac{c_{5\text{ф}} - V}{c_5 - V}. \tag{3.7}$$

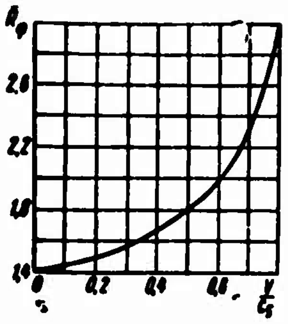


Fig. 3.15. The effect of parameter V/c_5 on the degree of thrust augmentation.

Let us assume that the degree of thrust augmentation of TRDF under stand conditions is equal to

$$\bar{R}_{\Phi_0}=\frac{c_{5\Phi_0}}{c_{5_0}}=\sqrt{\frac{T^*_{4\Phi}}{T^*_4}}. \qquad (3.8)$$

At all flight speeds and altitudes let us take $T^*_4=\text{const}$ and $T^*_{4\Phi}=\text{const}$ (since $T^*_3=\text{const}$ and $n=\text{const}$).

Then

$$\frac{c_{5\Phi}}{c_5}=\frac{c_{5\Phi_0}}{c_{5_0}}=\bar{R}_{\Phi_0}. \qquad (3.9)$$

Having replaced in equation (3.7) the value of $c_{\Phi 5}$ from equation (3.9), we obtain

$$\bar{R}_{\Phi}=\frac{\bar{R}_{\Phi_0}c_5-V}{c_5-V} \qquad (3.10)$$

or

$$\bar{R}_{\Phi}=\frac{\bar{R}_{\Phi_0}-\frac{V}{c_5}}{1-\frac{V}{c_5}}. \qquad (3.11)$$

From equation (3.11) it follows that with increase of flight speed the degree of thrust augmentation continuously rises from $\bar{R}_{\Phi_0}$ to ∞, since ratio V/c_5 increases from 0 to 1 (Fig. 3.15). With increase of flight altitude when V = const the degree of augmentation is continuously decreased, since ratio V/c_5 is lowered.

CHAPTER IV

OPERATIONAL CHARACTERISTICS OF DUCTED-FAN ENGINES

§ 1. Thermodynamic Fundamentals of Control of Ducted-Fan Engines

Control Elements and Controllable Parameters of Ducted-Fan Engines

The problem of control of ducted-fan engines is carrying out such a law of change of engine operating conditions on a stand and in flight, at which there is ensured the advantageous course of its basic characteristics — thrust, specific fuel consumption, excess turbine power, etc.

Operating conditions of ducted-fan engine are characterized generally by more independent variables than turbojet and turboprop engines. This determines the necessity of having additional control elements for handling the most advantageous distribution of the airflow rate and effective work between the ducts, realization of the prescribed law of heat addition (fuel consumption) in the secondary duct, etc. In accordance with this circumstance the ducted-fan engine has additional (as compared to TRD) controlling factors: fuel consumption in the secondary duct (G_T^{II}), stator angle of the compressor of the secondary duct (φ_{HA}^{II}), area of exit (throat) section of the jet nozzle of the secondary duct (f_S^{II}), etc., and also control elements (including automatic machines) corresponding to them.

The indicated controlling factors additionally determine the controllable physical conditions ducted-fan engine parameters: exit temperature from the combustion chamber of the secondary duct (T^{*II}_{ϕ}), work transmitted from the primary duct for driving the compressor of the secondary duct (L_{T_I}), and the parameter equivalent to it – distribution factor of effective work between ducts – x, and also the compression ratio in secondary duct compressor π_{KII}, revolutions of the secondary duct compressor, flow rate of air in the secondary duct G_{II}, etc., where the number of controlling factors is exactly equal to the number of independent controllable parameters.

Certain particular cases of combination of the controlling factors and the controllable parameters of different gas-turbine aircraft engines are given in Table 1.

Table 1. Controlling factors and controllable parameters of different types of engines.

Type of engine	Controlling factors	Controllable parameters
TRD	G_T, f_5	n, T^*_3
TRDF	G_T, G_{T_ϕ}, f_5	n, T^*_3, $T^*_{4\phi}$
TVD	G_T, φ_B	a) n, T^*_3 б) n_1, n_2
DTRD	G_T, f^I_5, f^{II}_5	a) n, T^*_3, $\pi^*_{K_{II}}$ б) n_1, n_2, T^*_3
DTRDF^{II}	G_T, $G^{II}_{T_\phi}$, f^I_5, f^{II}_5	n, $\pi^*_{K_{II}}$, T^*_3, T^{*II}_{ϕ}

From the table it is clear that the number of independent controllable parameters of a ducted-fan engine can be greater by one, two and more units than for other gas-turbine aircraft engines.

Let us note that the transition to a two-shaft engine design by itself still does not signify the introduction of an additional controllable parameter. With fixed geometry of a single-shaft ducted fan engine the only controllable parameter n (or T*) corresponds to the only controlling factor G_T. Another parameter T^*_3 (or n) is automatically uniquely connected with the first. In a two-shaft

ducted-fan engine, as before, the only controllable parameter, for example the number of revolutions of high-pressure stage, corresponds to controlling factor $G_т$. Parameters T_3^* and $n_{нд}$ are uniquely connected with $n_{вд}$.

Figure 4.1a shows a block diagram of control of single-shaft DTRDF^{II} with two independent regulators — fuel feed to the basic chamber and to afterburner.

The connection between controlling factors and controllable parameters of the engine in this diagram has the form:

$$G_т \to T_3^* \to n;\quad G_{т_ф} \to T_ф^{*II}.$$

Block diagram of the control of a two-shaft ducted-fan engine with two independent regulators for fuel feed and the jet nozzle of the secondary duct is shown on Fig. 4.1b.

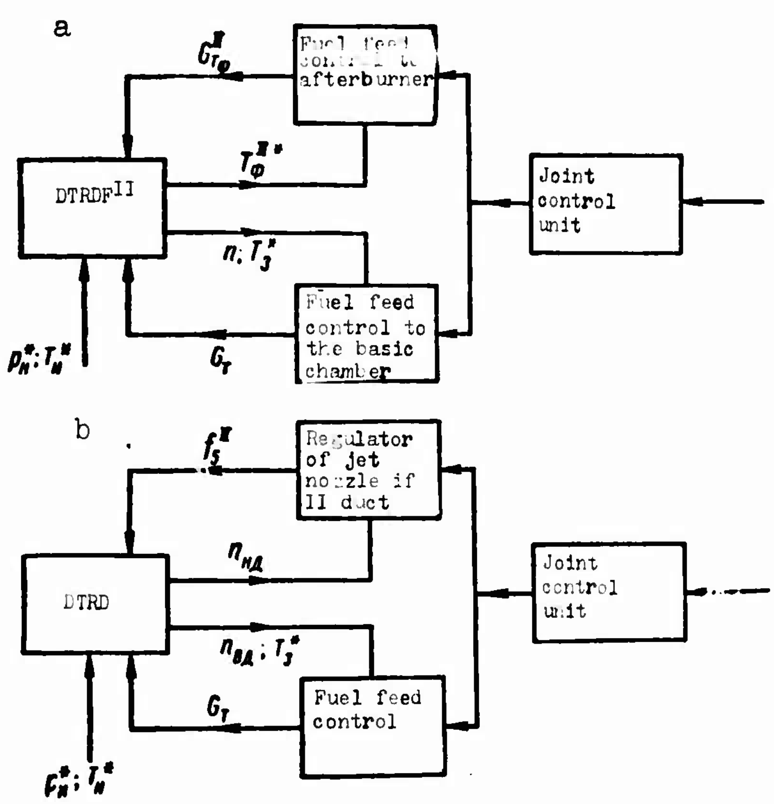


Fig. 4.1. Block diagram of control of DTRDF: a) with two regulators — fuel feed to basic chamber and afterburner; b) with two regulators — fuel feed and jet nozzle of the secondary duct.

The connection between controlling factors and controllable parameters of ducted-fan engine in the given diagram is the following:

$$G_{т} \rightarrow n_{вд} \rightarrow T_3^*; \quad f_5^{II} \rightarrow n_{нд}.$$

Programs of Control of Ducted-Fan Engines

Due to the presence of additional control elements (in the secondary duct of the engine) the amount of programs for a ducted-fan engine is considerably greater than for TRD. They include programs of control for maximum thrust, the best economy at flight cruise settings, complete similarity of operating conditions of the turbocompressor, different combined programs and others.

In this book there is no possibility of analyzing the shown programs. Let us briefly examine only the peculiarities of maximum thrust control of ducted-fan engines.

To provide maximum thrust of single-shaft ducted-fan engine at all flight speeds and altitudes it is necessary to realize the same conditions as for a single-shaft turbojet engine: $n=n_{макс}=\text{const}$, $T_3^*=T^*_{3_{макс}}=\text{const}$ and, furthermore, additional conditions $y=y_{опт}$, $x=x_{опт}$ or $\pi^*_{к}=\pi_{II_{опт}}$, and with the presence of afterburners in the ducts $T_{ф}^{I*}=\text{const}$ and $T_{ф}^{II*}=\text{const}$.

Observance of these conditions assures obtaining maximum airflow rate, specific thrust and consequently, maximum total thrust at all speeds and altitudes of flight.

It is necessary to note that provision of conditions $y=y_{опт}$, $x=x_{опт}$ or , $\pi^*_{кII}=\pi_{кII_{опт}}$ makes engine control very complicated and cannot always be realized practically. Therefore, usually we reject special control of parameters y and $\pi^*_{кII}$, allowing them to be changed in accordance with change of parameters of the working process with

respect to flight speed and altitude, and also conditions $n=n_{макс}=\text{const}$ and $T_3^*=T^*_{3_{макс}}=\text{const}$.

For ducted-fan engines, designed for supersonic flight speeds, it is expedient to control the optimum bypass ratio. In this case for the purpose of improving the economy of engine operation it is necessary to increase the bypass ratio on the stand and at subsonic flight speeds; at supersonic flight speeds conversely, it is expedient to decrease y (to 0), converting the ducted-fan engine to the operating conditions of a turbojet engine.

In a single-shaft ducted-fan engine $T_3^*=\text{const}$ can be maintained by additional control of the jet nozzle throat (of the primary or secondary ducts, and with the presence of a mixing chamber — common nozzle), and also of the stator of the secondary duct compressor.

In connection with this, additional conditions of realizing the examined program of control distinguished:

$$f_5^{II}=\text{const},\quad \varphi_{HA}^{II}=\text{const}\ (f_5^{I}=\text{var});$$
$$f_5^{I}=\text{const},\quad \varphi_{HA}^{II}=\text{const}\ (f_5^{II}=\text{var});$$
$$f_5^{I}=\text{const},\quad f_5^{II}=\text{const}\ (\varphi_{HA}=\text{var}).$$

In a two-shaft ducted-fan engine $T_3^*=\text{const}$ is preserved automatically with the observance of condition $n_{вд}=\text{const}$ with accuracy up to $L_{квд}=\text{const}$ in relation to the speed and altitude of flight. Preservation of $L_{квд}=\text{const}$ can be assured by corresponding selection of the characteristic of a high-pressure compressor with rated compression ratio close to 6.

With observance of condition $n_{нд}$ = const and fixed geometry of the engine the temperature T_3^* is changed in flight. For maintaining it the introduction of earlier considered additional control elements is required.

Combined Work of Compressor and Turbine in a Ducted-Fan Engine.
The Effect of Various Controlling Facotrs on the Line
of Operating Conditions of DTRD Compressors

The course of throttle, altitude and high-speed characteristics of ducted-fan engines, selection of programs of control for their realization are determined to a great extent by peculiarities of the combined work of the compressor and turbine, peculiarities of the gas-dynamic circuit of the engine, depend on the properties and actual characteristics of compressors of the primary and secondary ducts.

Let us examine methods of construction of the line of operating conditions of DTRD compressors, and the effect of different operational and control factors on their course.

Basic Conditions and Assumptions.

During construction of lines of operating conditions on the characteristics of DTRD compressors of different circuits let us proceed from the following conditions and assumptions.

1. Pressure drops in the primary nozzle box of the turbine, and also in jet nozzles of both ducts have critical or supercritical values, i.e.,

$$q(\lambda^{I}_{CA}) = 1, \quad q(\lambda^{I}_{5}) = 1, \quad q(\lambda^{II}_{5}) = 1.$$

This means that the expansion ratio of gas in the turbine remains constant, π^{*}_{T} = const.

On Fig. 4.2 there are listed dependencies of the maximum bypass ratio (y) on parameters of the working process of the primary duct (T^{*}_{3} and π^{*}_{K}), at which the pressure drops in both ducts reach critical values. When $y < 2$, $T^{*}_{3} > 1300°$ and $\pi^{*}_{K} > 10$, and also $\eta^{*}_{T} = 0{,}9$ and $\eta^{*}_{K} = 0{,}85$ we have

$$\pi_{p.c_{I}} = \pi_{p.c_{II}} > \pi_{kp} = \left(\frac{k-1}{2}\right)^{\frac{k}{k-1}}.$$

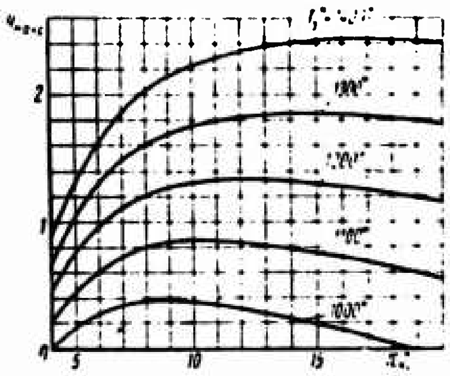


Fig. 4.2. The effect of parameters of the DTRD working process on $y_{нскр}$ at critical outflow of gas from the jet nozzles ($M_0=0,\ H=0,\ \pi_{р.с_I}=\pi_{р.с_{II}}=\pi_{кр}$).

2. Efficiencies of the turbine, and also loss factors in the engine components are constant:

$$\eta_т^* = \text{const},\quad \sigma_{вх}^* = \text{const},\quad \sigma_{к.с}^* = \text{const},\quad \varphi_{р.с} = \text{const},\quad \xi_{к.с} = \text{const},\quad \sigma_{ф.к}^* = \text{const};\quad \xi_{ф.к} = \text{const}.$$

Single-Shaft Ducted-Fan Engine with Separate Gas-Air Ducts (See Fig. 1.3a)

1. Equations of Basic Gas-Dynamic Connections of the Engine Turbocompressor (Equations of Conditions of the Joint Work of Turbine and Compressor).

In reference to the examined engine diagram the equation of conditions of the joint work of compressor and turbine include equations of the flow rate of air (gas) through the primary and secondary engine ducts, and also equation of the balance of works (powers) of the turbocompressor of the engine.

Equation of the Flow Rate of Air Through the Primary Duct.

Let us write the equation of flow rate of air (gas) for the compressor inlet section 1-1 and the throat of the primary nozzle box of the turbine

$$m_в \frac{p_1^*}{\sqrt{T_1^*}} q(\lambda_1) f_1 = m_г \frac{p_{CA}^*}{\sqrt{T_{CA}^*}} q(\lambda_{CA}) f_{CA}. \tag{4.1}$$

whence

$$\pi^{*}_{\kappa_{I}} = \frac{q(\lambda^{I}_{1})}{q(\lambda_{CA})} \cdot \frac{f^{I}_{1}}{f_{CA}} \cdot \frac{\bar{m}}{\sigma^{*}_{\kappa.c}\,\sigma^{*}_{CA}} \sqrt{\frac{T^{*}_{3}}{T^{*}_{H}}}, \tag{4.2}$$

or for fixed-area sections of the primary duct

$$\pi^{*}_{\kappa_{I}} = c_{1} q(\lambda^{I}_{1}) \sqrt{\frac{T^{*}_{3}}{T^{*}_{H}}}, \tag{4.3}$$

where

$$c_{1} = \frac{f^{I}_{1}}{f_{CA}} \cdot \frac{\bar{m}}{\sigma^{*}_{\kappa.c}\,\sigma^{*}_{CA}}; \quad q(\lambda_{CA}) = 1; \quad \bar{m} = \frac{m_{B}}{m_{r}}.$$

Equation (4.3) does not differ from the analogous equation for a single-duct turbojet engine.

Equation of the Flow Rate of Air Through the Secondary Duct.

Let us write the equation of flow rate of air for the compressor inlet section 1-1 and the throat of the jet nozzle for the secondary duct 5^{II}-5^{II}

$$m_{B} \frac{p^{*}_{1}}{\sqrt{T^{*}_{1}}} q(\lambda^{II}_{1}) f^{II}_{1} = m_{r} \frac{p^{*II}_{5}}{T^{*II}_{5}} q(\lambda^{II}_{5}) f^{II}_{5}, \tag{4.4}$$

whence

$$\pi^{*}_{\kappa_{II}} = \frac{q(\lambda^{II}_{1}) f^{II}_{1} \bar{m}}{q(\lambda^{II}_{5}) f^{II}_{5}\, \sigma^{*II}_{\kappa.c}} \sqrt{\frac{T^{*II}_{5}}{T^{*}_{H}}}. \tag{4.5}$$

In a particular case, when fuel is not burned in the secondary duct of the engine $(T^{*II}_{5} = T^{*II}_{2})$, equation (4.5) takes the form:[1]

$$\pi^{*\frac{n+1}{2n}}_{\kappa_{II}} = \frac{\pi^{*}_{\kappa_{II}}}{\sqrt{1 + \dfrac{(\pi^{*0,286}_{\kappa_{II}} - 1)}{\eta^{*}_{\kappa_{II}}}}} = \frac{q(\lambda^{II}_{1}) f^{II}_{1}}{q(\lambda^{II}_{5}) f^{II}_{5}\, \sigma^{*II}_{\kappa.c}}. \tag{4.6}$$

[1]Bearing in mind that $\dfrac{T^{*II}_{2}}{T^{*}_{1}} = \pi^{*\frac{n-1}{n}}_{\kappa_{II}} = 1 + \dfrac{(\pi^{*0,286}_{\kappa_{II}} - 1)}{\eta^{*}_{\kappa_{II}}}$.

and with fixed section of the secondary duct and critical exhaust velocities from it

$$\pi_{\kappa_{II}}^{*\frac{n+1}{2n}} = c_2 q(\lambda_1^{II}), \tag{4.7}$$

where

$$c_2 = \frac{f_1^{II}}{f_5^{II}\sigma_{\kappa.c}^{*II}}; \quad q(\lambda_5^{II}) = 1;$$

n — polytropic exponent of compression in the compressor.

The connection between polytropic exponent and the efficiency of compressor is determined by formula

$$\eta_\kappa^* = \frac{\frac{k-1}{k}}{\frac{n-1}{n}}.$$

When $\eta_\kappa^* = 0.85$ we have n = 1.5 and $\frac{2n}{n+1} = 1.2$.

Equation of Work Balance of the Turbocompressor.

We have

$$L_{\kappa_I} + yL_{\kappa_{II}} = L_T. \tag{4.8}$$

This equation can be written in the form:

$$\frac{c_{p_r}}{A} T_3^* \left(1 - \frac{1}{e_T^*}\right) \eta_T^* = L_{\kappa_I} + yL_{\kappa_{II}}. \tag{4.9}$$

Here

$$y = \frac{G_{II}}{G_I} = \frac{G_5^{II}}{G_{CA}} = \frac{\pi_{\kappa_{II}}^*}{\pi_{\kappa_I}^*} \sqrt{\frac{T_3^*}{T_5^{*II}}} \cdot \frac{f_5^{II} q(\lambda_5^{II})}{f_{CA} q(\lambda_{CA})} \cdot \frac{\sigma_{\kappa.c}^{*II}}{\sigma_{\kappa.c}^* \sigma_{CA}^*}. \tag{4.10}$$

[illegible]. Peculiarities of Lines of the Operating Conditions of the Compressor of the Secondary Duct.

Flow rate equations (4.5) and (4.6) give the possibility of constructing lines of operating conditions on the characteristic of the compressor of the secondary duct (Fig. 4.3). Let us examine several particular cases.

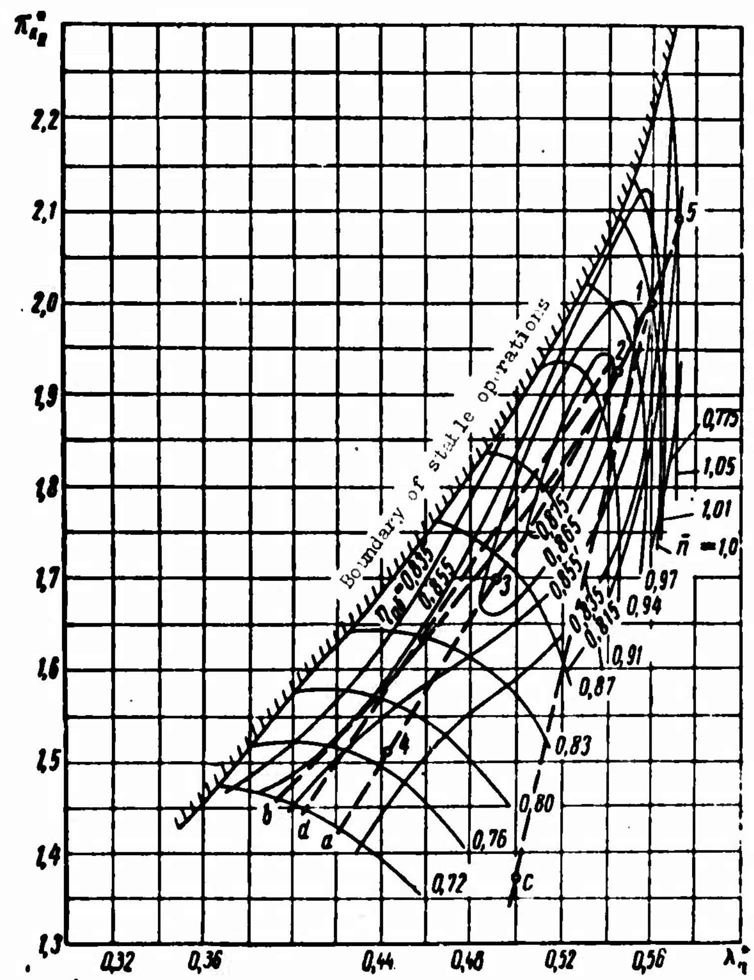


Fig. 4.3. Characteristic of the secondary duct of a ducted-fan engine with lines of operating conditions: 1 – $M_0 = 0$, $H = 0$; 2 – $M_0 = 1.5$, $H = 11$ km; 3 – $M_0 = 2$, $H = 11$ km; 4 – $M_0 = 2.5$, $H = 11$ km; 5 – $M_0 = 0.9$, $H = 11$ km.

Program of Control n = const, f^{II}_{IIA} = const, f^{II}_{5} = const (Secondary Duct Is Geometrically Fixed).

At a critical pressure drop in the jet nozzle $(q(\lambda^{II}_{5}) = 1)$ the line of operating conditions [see equation (4.6)] is depicted by slightly curved parabola 1-a. If, however, we consider the subcritical pressure drops $(q(\lambda^{II}_{5}) < 1)$, which set in during engine throttling, then the line of operating conditions 1-b will be deflected to the region of smaller values of $q(\lambda^{II}_{1})$.

From equation (4.5) it follows that a specified line of operating conditions corresponds to the value of the throat area of the nozzle f^{II}_{5}. Analogous to the primary duct of turbojet engine (ducted-fan engine) it is possible to construct a family of line of operating conditions f^{II}_{5} = const on the characteristic of compressor of the secondary duct. With decrease of f^{II}_{5} the LRR of the secondary duct is displaced to the region of raised values of $\pi^{*}_{K_{II}}$.

It is characteristic that in the system of relative coordinates $\bar{\pi}^{*}_{K_{II}} = f[\bar{q}(\lambda^{II}_{1})]$ the lines of operating conditions, constructed for different computed values of $\pi^{*}_{K_{II}}$, coincide.

Program of Control n = const, f^{II}_{5} = const, T^{*II}_{Φ} = const.

At constant reheat temperature in the secondary duct equation (4.5) takes the form:

$$\pi^{*}_{K_{II}} = \frac{C}{\sqrt{T^{*}_{H}}} q(\lambda^{II}_{1}), \tag{4.11}$$

where C = const.

Line of operating conditions 1-c, constructed by this equation, is considerably steeper than in the case of the nonreheated secondary duct. Really, increase of gas temperature past the fan $T^{*II}_{2} = T^{*II}_{5}$ with rise of flight speed acts as a thermal throttle in the nozzle throat and accelerates the drop of $q(\lambda^{II}_{1})$. Preservation of $T^{*II}_{5} = T^{*II}_{\Phi}$ = const delays the lowering of $q(\lambda^{II}_{1})$.

In the case when the jet nozzle is controlled according to the law

$$f_5^{II} \sim \sqrt{T_\Phi^{*II}} = \text{const},$$

the line of operating conditions of the compressor of the secondary duct coincides with curve 1-c.

Program of Control n = const, $T^{*II}_{\Phi_{np}}$ = const, f_5^{II} = const.

If the cited reheat temperature is kept constant, equation (4.5) takes the form:

$$\pi_\kappa^{*II} = C q(\lambda_1^{II}) \qquad (4.12)$$

and is depicted on Fig. 4.3 by straight line 1-d.

3. Peculiarities of Lines of Operating Conditions of the Compressor of the Primary Duct.

Equations of flow rate (4.3) and work balance (4.8) give the possibility of constructing a line of operating conditions on the characteristic of the compressor of the primary duct.

Program of Control n = const, f_5^{I} = const, f_5^{II} = const, φ_{HA} = const.

Figure 4.4 shows line of operating conditions 5'-1-4' of the compressor of the primary duct of a geometrically fixed ducted-fan engine. It is flatter than the line of operating conditions of a geometrically fixed turbojet engine. This means that with equal values of the given revolutions the higher values of gas temperature before the turbine corresponds to the line of operating conditions of ducted-fan engine. Inevitable increase of the bypass ratio with decrease of the given revolutions causes increase of temperature T_3^* [see equations (4.9) and (4.10)]. The higher the value of $y_{\text{расч}}$, the more sharply T* rises with increase of flight speed and the flatter, i.e., with a smaller surge margin of stability, is the line of operating conditions of ducted-fan engine (Fig. 4.5).

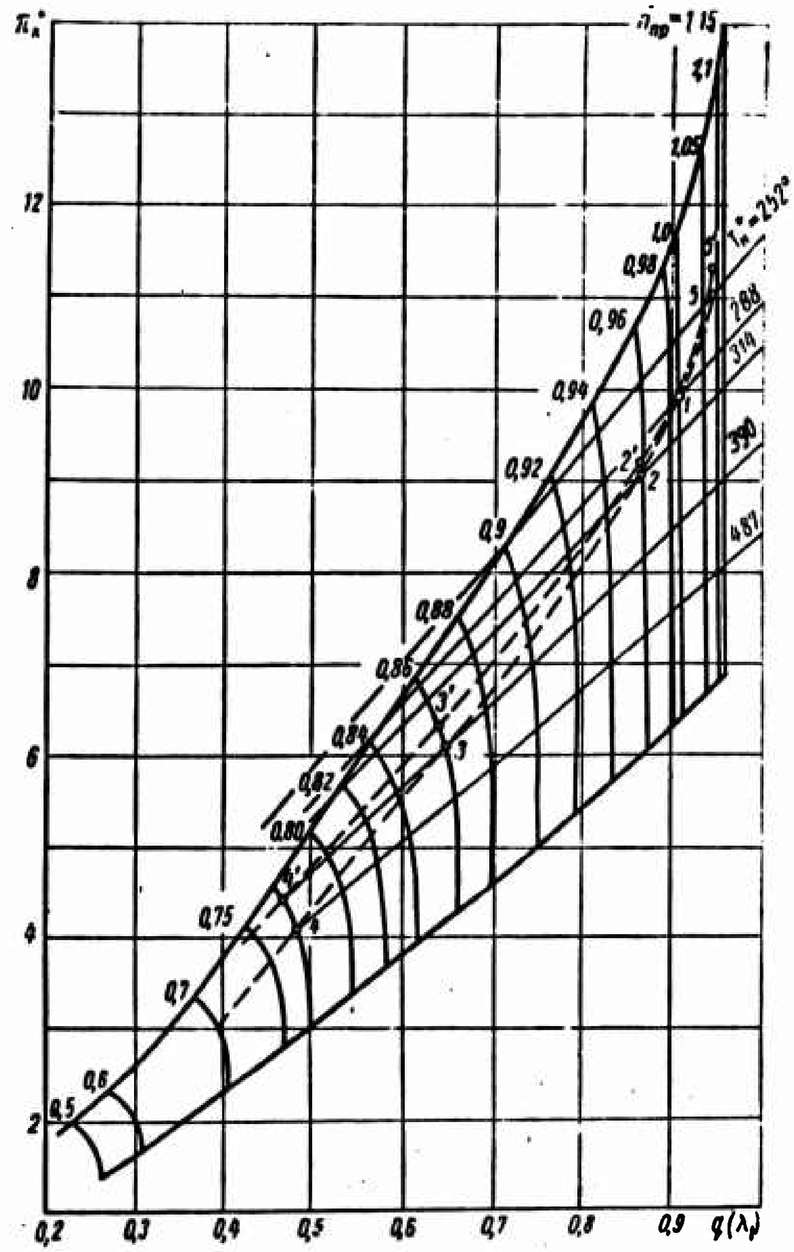


Fig. 4.4. Characteristic of the primary duct of ducted-fan engine with lines of operating conditions.

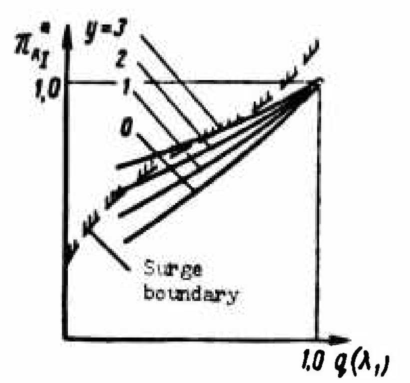


Fig. 4.5. The effect of the bypass ratio on the IRR of the primary duct of ducted-fan engine.

Program of Control n = const, $T_3^* =$ const.

On Fig. 4.4 there is drawn line of operating conditions 5-1-4, constructed for the examined program with the presence of variable-area nozzle of the primary duct $(f_5^{\text{I}}-\text{var})$. Lines of operating conditions n = const and T_3^* = const for turbojet and ducted-fan engine completely coincide, regardless of what control elements were used to assure constancy of the gas temperature before the turbine $(f_5^{\text{I}}$ = var or f_8^{II} = var, etc.).

Two-Shaft Ducted-Fan Engine with a Common Low-Pressure Compressor and Separate Exhaust (See Fig. 1.3b)

1. Equations of Basic Gas-Dynamic Connections.

Equations of basic gas-dynamic connections of ducted-fan engines of the given diagram include equations of flow rate and work balance of low-pressure and high-pressure turbocompressors.

High-Pressure Turbocompressor.

Flow Rate Equation.

By analogy with TRD let us write

$$\pi^*_{\text{к}_{\text{ВД}}} = c_1 q(\lambda_1)_{\text{ВД}} \sqrt{\frac{T_3^*}{T^*_{1(\text{ВД})}}}. \qquad (4.13)$$

where

$$c_1 = \frac{f_{1(\text{ВД})}}{f_{\text{CA}} q(\lambda_{\text{CA}}) \sigma^*_{\text{CA}} \sigma^{*\text{I}}_{\text{к.c}}}.$$

Equation of Work Balance.

We have

$$L_{\text{ТВД}} = L_{\text{КВД}}. \qquad (4.14)$$

Whence we find

$$\frac{T_3^*}{T_{1(ВД)}^*} = \frac{102.5\left(\pi_{к_{ВД}}^{*0.286} - 1\right)\frac{1}{\eta_{к_{ВД}}^*}}{118\,\varepsilon_{т_{ВД}}^*\eta_{т_{ВД}}^*}. \tag{4.15}$$

Having placed the value of $\frac{T_3^*}{T_{1_{ВД}}^*}$ from formula (4.15) into equation (4.13), after conversions we obtain

$$\frac{\pi_{к_{ВД}}^*}{\sqrt{\dfrac{\pi_{к_{ВД}}^{*0.286} - 1}{\eta_{к_{ВД}}^*}}} = cq(\lambda_1)_{ВД}, \tag{4.16}$$

where

$$c = \frac{f_{1_{ВД}}}{f_{СА}\,\sigma_{СА}^*\,\sigma_{к.с}^{*1}\sqrt{\dfrac{118}{102.5}\,\varepsilon_{т_{ВД}}^*\,\eta_{т_{ВД}}^*}}.$$

Equation (4.16) is the equation of lines of operating conditions of the high-pressure turbocompressor. It is analogous to a similar equation, formulated for the high-pressure turbocompressor of a single-spool turbojet engine.

Low-Pressure Turbocompressor.

Flow Rate Equation.

Let us write the flow rate equation for inlet section of low-pressure compressor 1-1 and exit section of the convergent jet nozzle of the secondary duct 5^{II}-5^{II}:

$$G_{1_{НД}} = G_5^{II}\left(1 + \frac{1}{y}\right) \tag{4.17}$$

or

$$m\frac{p^*_{1(НД)}}{\sqrt{T^*_{1(НД)}}}f_{1НД}\,q(\lambda_1)_{НД}=m\frac{p_5^{*II}}{\sqrt{T_5^{*II}}}f_5^{II}q(\lambda_5^{II})\left(1+\frac{1}{y}\right).$$

Bearing in mind that $T_5^*=T^*_{2(НД)}$ and $p_5^{*II}=p_2^{*II}\sigma^{*II}_{2-5}$, after conversions we obtain

$$\pi^{*\frac{n+1}{2n}}_{кНД}=\frac{\pi^*_{кНД}}{1+\dfrac{\pi^{*0.286}_{кНД}-1}{\eta^*_{кНД}}}=\frac{c_2}{1+\dfrac{1}{y}}q(\lambda_1)_{НД}, \tag{4.18}$$

where

$$c_2=\frac{f_{1(НД)}}{f_5^{II}\sigma^{*II}_{2-5}}.$$

Equation of Work Balance.

We have

$$G_I L_{тНД}=(G_I+G_{II})L_{кНД},$$

whence

$$L_{тНД}=(1+y)L_{кНД}. \tag{4.19}$$

Expression of Bypass Ratio of DTRD.

We have

$$y=\frac{G_5^{II}}{G_{1(ВД)}}=\frac{\dfrac{p_5^{*II}}{\sqrt{T_5^{*II}}}f_5^{II}q(\lambda_5^{II})}{\dfrac{p^*_{1(ВД)}}{\sqrt{T^*_{1(ВД)}}}f_{1(ВД)}q(\lambda_1)_{ВД}}.$$

Taking into account that $p_5^{*II}=p^*_{1(ВД)}\sigma^{*II}_{2-5}$ and $T_5^{*II}=T^*_{1(ВД)}$, after simplifications we obtain

$$y = \frac{\sigma^{*II}_{2-5} f^{II}_5 q(\lambda^{II}_5)}{f_{1(\text{ВД})} q(\lambda_1)_{\text{ВД}}}. \tag{4.20}$$

Thus, change of the bypass ratio of a geometrically fixed ducted-fan engine of the examined diagram is determined only by change of the flow rate parameter $q(\lambda_1)_{\text{ВД}}$. Consequently, having constructed the LRR of high-pressure compressor, it is possible for any conditions to determine the new values of the bypass ratio (Fig. 4.6) by formula

$$y = \frac{y_{\text{расч}}}{\bar{q}(\lambda_1)_{\text{ВД}}}, \tag{4.21}$$

where

$$\bar{q}(\lambda_1)_{\text{ВД}} = \frac{q(\lambda_1)_{\text{ВД}}}{q(\lambda_1)_{\text{ВД}_{\text{расч}}}}.$$

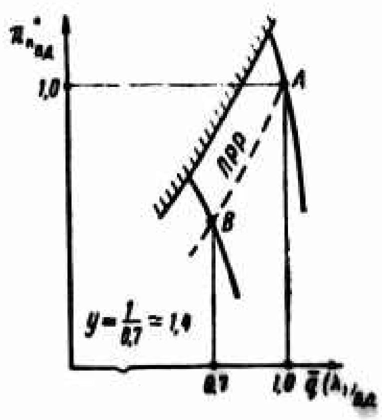


Fig. 4.6. Graphic determination of y by characteristic of high-pressure compressor (characteristic is given in relative parameters).

By using equations (4.13), (4.16), (4.18), (4.19) and (4.20), it is possible to construct lines of operating conditions on the characteristics of high- and low-pressure compressors for two of the most often encountered control programs of geometrically fixed ducted-fan engines:

1) $n_{\text{КВД}} = \text{const}$, $f^{I}_5 = \text{const}$, $f^{II}_5 = \text{const}$, $\varphi_{\text{НА}} = \text{const}$;

2) $n_{\text{КНД}} = \text{const}$, $f^{I}_5 = \text{const}$, $f^{II}_5 = \text{const}$, $\varphi_{\text{НА}} = \text{const}$.

Comparison of Lines of Operating Conditions on the Characteristics of High-Pressure and Low-Pressure Compressors with Two Programs of Control.

By comparing performance points, located along the lines of operating conditions on characteristics of high-pressure and low-pressure compressors, at the two above-indicated programs of control (4.7), the following conclusions can be made.

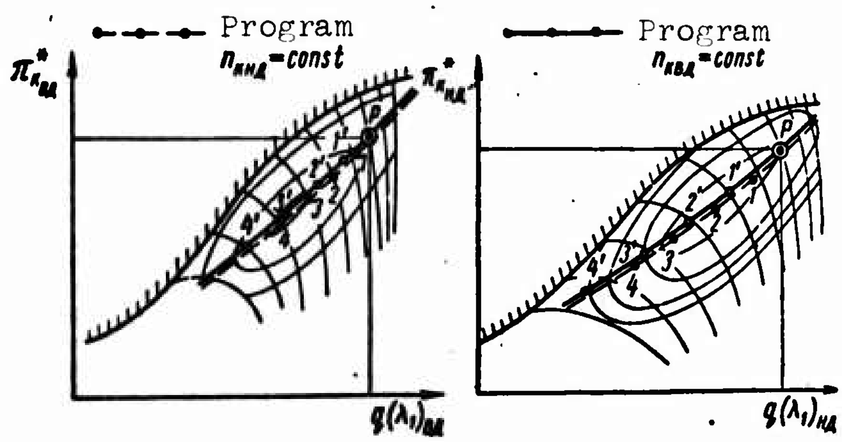


Fig. 4.7. The effect of the program of control on the course of LRR on the characteristics of high- and low-pressure compressors of two-shaft ducted-fan engines.

1. Lines of operating conditions on the characteristic of each compressor at the given programs of control coincide.

2. For both programs of control the relative change of giv revolutions of the high-pressure compressor is less than the giv revolutions of the low-pressure compressor. This is explained by the fact that in the prescribed interval of changes of $\bar{T}^*_{Н} = \bar{T}^*_{1(НД)}$ the change of parameter $\bar{T}^*_{1(ВД)}$ is always smaller.

3. With increase of $T^*_{Н}$ at program of control $n_{КВД}$ = const the revolutions of the free low-pressure stage are reduced $[(1+y)L_{КНД} > L_{ТНД}]$, and at program of control $n_{КНД}$ = const the revolutions of the free high-pressure stage increase (to assure $n_{НД}$ = const it is necessary to increase T^*_3, and consequently, $L_{ТВД} > L_{КВД}$).

Thus, the larger change of given revolutions of high- and low-pressure compressors at program of control $n_{КВД}$ = const and their smaller change at program of control $n_{КНД}$ = const correspond to the same interval of changes of T_H^*. This follows from analysis of formulas

$$n_{ВД_{пр}} = n_{ВД}\sqrt{\frac{288}{T^*_{1(ВД)}}} \quad \text{and} \quad n_{НД_{пр}} = n_{НД}\sqrt{\frac{288}{T_H^*}}.$$

4. A larger change of given revolutions causes a larger deviation of compressor parameters ($L_к$, $\eta_к^*$, $\pi_к^*$) from their initial (rated) values. In this sense the program of control $n_{КНД}$ = const is more preferable. However, T_3^* is changed less in the case $n_{КВД}$ = const.

Physical Model of the Change of Bypass Ratio in Relation to the Speed and Altitude of Flight

Let us examine two typical cases of change of the bypass ratio depending on the speed and altitude of flight (Fig. 4.8):

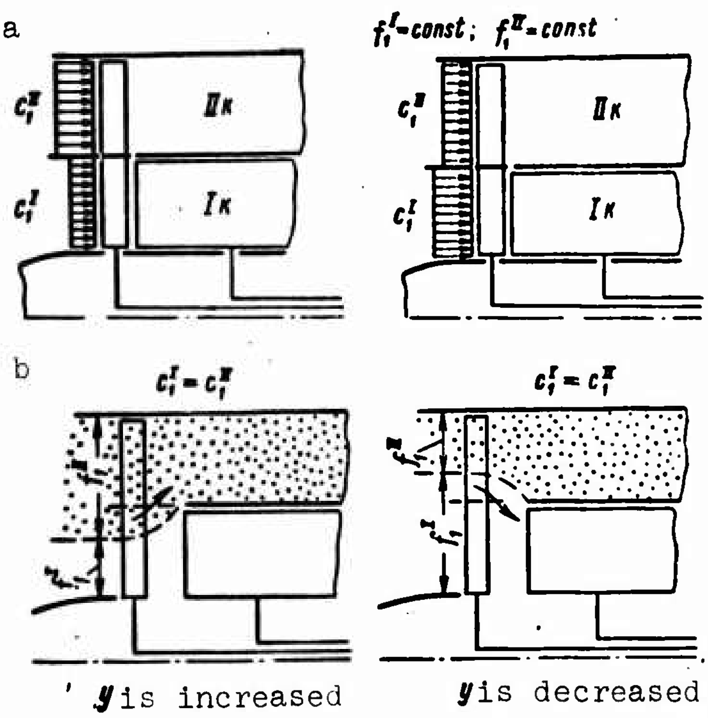


Fig. 4.8. Physical model of change of the bypass ratio with respect to the speed and altitude of flight: a) redistribution of profile of axial velocities at the low-pressure compressor inlet (blade with delimitation flanges); b) deformation of the stream at the low-pressure compressor inlet (common blades).

with the presence of delimitation flanges on blades of the low-pressure compressor (diagram of ducted-fan engine with separation the airflow at the compressor inlet);

with the presence of common low-pressure compressor blades (without delimitation flanges).

In the first case the bypass ratio of DTRD depending on the speed and altitude of flight is changed due to redistribution of the profile of axial velocities at the compressor inlet. For example, with increase of T_H^* (and consequently, parameter y) the inlet velocity at the compressor of the secondary duct increases (Fig. 4.8a); conversely, with decrease of y the inlet velocity at the compressor of the primary duct increases.

In the second case the bypass ratio is changed due to deformation of the stream at the compressor inlet (Fig. 4.8b). Thus, for example, with increase of parameter y the cross section of the stream flowing into the secondary duct (f_I^{II}); increases; accordingly, the stream at the primary duct compressor inlet (f_I^{I}) is narrowed. With decrease of parameter y, conversely, the stream at the inlet of the compressor of the secondary duct is narrowed and at the inlet of the compressor of the primary duct is expanded.

§ 2. Throttle Characteristics of Ducted-Fan Engines

The dependence of total thrust and specific fuel consumption on the number of revolutions of the turbocompressor (or position of the control elements of the engine) at constant flight speed and altitude and with the accepted program of control is called the throttle characteristic of ducted-fan engine.

On the throttle characteristic of a ducted-fan engine, as in the case of turbojet engine, there are plotted curves of change of the gas temperature in the jet nozzle (past the turbine), consumption of fuel per hour, and also other characteristic dependencies for ducted-fan engines (for example, y = f(n), π_{KII}^* = f(n) and others.

Throttle characteristic of ducted-fan engine at prescribed computed values of parameters of the working process of the primary duct (T_3^*, $\pi_{\kappa I}^*$) to a considerable degree depends on the gas-dynamic circuit of the engine [(DTRD(P) or DTRD(Z), single- or two-shaft engine], its control system and characteristic parameters, determining the distribution of air and energy between the ducts.

Throttle Characteristic of Single-Shaft DTRD

Change of Pressure Drops in the Turbine and Jet Nozzle of the Primary Duct of Single-Shaft Ducted-Fan Engines During Throttling.

With decrease of engine revolutions the compression ratio of the compressor of the primary duct is lowered approximately the same as for an initial single-spool turbojet engine. Difference of dependencies $\pi_\kappa^* = f(n)$ for ducted-fan and turbojet engine is determined by peculiarities of the course of lines of operating conditions on the compressor characteristic.

Earlier we clarified that the expansion ratio of the DTRD turbine is considerably greater than for the initial TRD. Therefore, the pressure drop in the jet nozzle of the primary duct of the DTRD is always smaller than for the initial TRD, i.e., $\pi_{p.c(I)} < \pi_{p.c(0)}$.

Let us consider a case when at maximum engine operating conditions the pressure drop in the jet nozzle is subcritical, i.e.,

$$\pi_{p.c(I)} < \pi_{\kappa p} = \left(\frac{k+1}{2}\right)^{\frac{k}{k-1}} = 1{,}85.$$

Comparatively low values of T_3^* and high values of y correspond to this case (Fig. 4.2). Then, when throttling the engine as the revolutions and, consequently, the compression ratio of the compressor lower, the pressure drop in the jet nozzle and the turbine is decreased. At first the lowering of total expansion ratio mainly encompasses the low-pressure stage — jet nozzle. Then as $\pi_{p.c(I)}$ approaches 1 the fall of pressure drop in the turbine is intensified (Fig. 4.9).

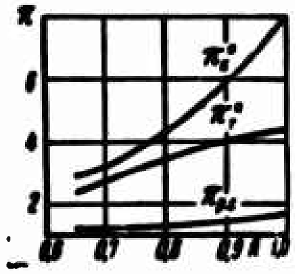


Fig. 4.9. Redistribution of pressure drop between the turbine and jet nozzle when throttling a single-shaft ducted-fan engine ($T^{*}_{3(p)}$ = 1200°K, y_p = 2, $\pi^{*}_{\kappa I(p)}$ = 8, $\pi^{*}_{\kappa II(p)}$ = 1.7).

Thus, the regularity of change of pressure drop in the DTRD turbine noticeably differs from the change of pressure drop in the turbine of a single-spool turbojet engine, for which the pressure drop in the turbine remains constant in a considerable range of revolutions.

Distribution of the Total Expansion Ratio Between the Turbine and Jet Nozzle During Engine Throttling.

Distribution of the total expansion ratio between the turbine and jet nozzle under subcritical conditions of outflow of gas from the jet nozzle can be accomplished by the method discussed for turbojet engines. On Fig. 4.10 there is shown the distribution of total expansion ratio of gas between the turbine and jet nozzle during engine throttling. Change of $\pi^{*}_{\kappa I}$, π^{*}_{T}, and $\pi_{p.c}$ with respect to the number of revolutions for a single-shaft ducted-fan engine are shown in Fig. 4.9.

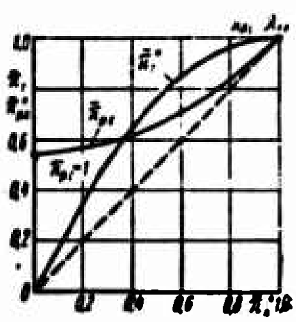


Fig. 4.10. Distribution of the total expansion ratio of gas between the turbine and jet nozzle during throttling $\left(\eta^{*}_{T}=1,\ k=\frac{4}{3}\right)$.

Change of the Bypass Ratio of a Single-Shaft Ducted-Fan Engine During Throttling.

With decrease of the number of revolutions of a geometrically fixed single-shaft ducted-fan engine, its bypass ratio is at first increased (Fig. 4.11). Such regularity is explained by the fact that the mass flow rate of air through the given duct depends mainly on the compression ratio of the compressor. During engine throttling the compression ratio in the primary duct, having a higher initial value, is decreased more intensively than in the secondary duct. This leads to the fact that the flow rate of air through the primary duct drops much more intensely than through the secondary duct, as a result of which the bypass ratio is increased.

$$y=\frac{G_{II}}{G_I}=\frac{\pi^*_{K_{II}}}{\pi^*_{K_I}}\sqrt{\frac{T^*_3}{T^{*II}_3}}\cdot\frac{f^{II}_5}{f_{CA}}\cdot\frac{q(\lambda^{II}_5)}{q(\lambda^{I}_{CA})}\approx\frac{\pi^*_{K_{II}}}{\pi^*_{K_I}}$$

Only in the area of rough throttling, when the gas exit velocity from the secondary duct is lowered, does y drop.

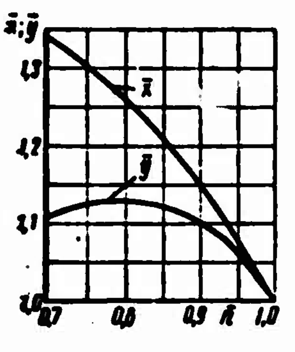


Fig. 4.11. Change of bypass parameters of a single-shaft ducted-fan engine during throttling.

Increase of parameter y with reduction of the number of revolutions is an important factor, which, as we will see further, seriously affects the operational features and properties of ducted-fan engines.

Change of Gas Temperature Before the Turbine of a Single-Shaft Ducted-Fan Engine.

The gas temperature before turbine of a single-shaft ducted-fan engine is determined by the balance of works of turbine and compressor at equilibrium revolutions. We have

$$L_{\text{т}}=L_{\text{к}_{\text{I}}}+yL_{\text{к}_{\text{II}}}=\frac{c_{p_{\text{г}}}}{A}T_3^*e_{\text{т}}^*\eta_{\text{т}}^*,$$

whence

$$T_3^*=\frac{L_{\text{к}_{\text{I}}}+yL_{\text{к}_{\text{II}}}}{\frac{c_{p_{\text{г}}}}{A}e_{\text{т}}^*\eta_{\text{т}}^*}, \qquad (4.22)$$

where

$$e_{\text{т}}^*=1-\frac{1}{\pi_{\text{т}}^{*0.25}}; \quad \frac{c_{p_{\text{г}}}}{A}=118.$$

Let us assume, as before, that at initial maximum operating conditions of the engine $\pi_{\text{р.с(I)}}<\pi_{\text{кр}}$. With decrease of the number of revolutions the compressor work will be continuously lowered. If the pressure drop in the turbine and the bypass ratio would remain constant in a certain range of revolutions, the gas temperature before the turbine of a ducted-fan engine would be ilitially lowered in the same way as for a TRD. But since parameter y increases during throttling (compressor is "loaded", compressing a relatively greater quantity of air in the secondary duct), and the pressure drop on the turbine is rapidly lowered (turbine is "lightened", its efficiency drops), then as a result the gas temperature before the turbine of a ducted-fan engine is decreased insignificantly as compared to a turbojet engine. With further engine throttling the temperature, having reached a certain minimum, starts to rapidly rise. In the entire range of working revolutions the value of T_3^* for derivative of DTRD turns out to be substantially higher than for the initial turbojet engine (Fig. 4.12). Fig. 4.12 shows that with lowering of engine revolutions by 35%, the relative value of $\bar{T}_3^*$

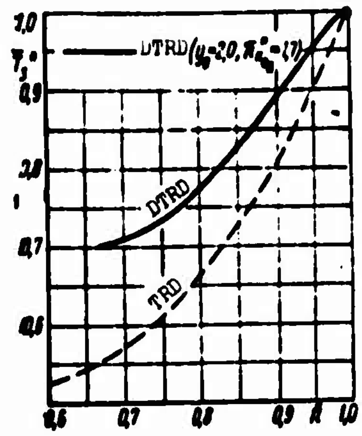


Fig. 4.12. Change of the relative gas temperature before the turbine of a single-shaft ducted-fan engine with respect to the number of revolutions ($H=0$, $c_0=0$, $T_3^*=1200°$ K, $\pi^*_{k\,I(0)}=8$, $\eta_k^*=0.85$, $\eta_T^*=0.9$).

for ducted-fan engine is equal to 0.70, and for turbojet engine — 0.55; thus, the gas temperature before the turbine of the ducted-fan engine turns out to be 190° higher.

The effect of the bypass ratio on the relative change of gas temperature before the DTRD turbine during engine throttling is shown in Fig. 4.13. The larger the value of y, and consequently, the lower is $\pi^*_{k_{II}}$, the higher the level of temperature $\bar{T}_3^*$ in the entire range of engine working revolutions.

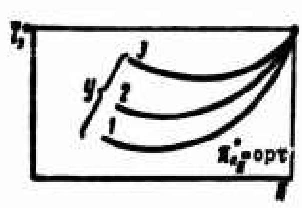


Fig. 4.13. Effect of the bypass ratio on the relative change of $\bar{T}_3^*$ during throttling of single-shaft ducted-fan engine.

Change of Thrust and Specific Fuel Consumption of Single-Shaft Ducted-Fan Engine.

Figure 4.14 shows the relative change of thrust and specific fuel consumption of a single-shaft ducted-fan engine depending on the number of revolutions.

In accordance with the higher values of gas temperature before the turbine and the delayed drop of the airflow rate during throttling, the DTRD thrust drops slower than for a turbojet engine, and specific fuel consumption is increased faster, Thus, for example,

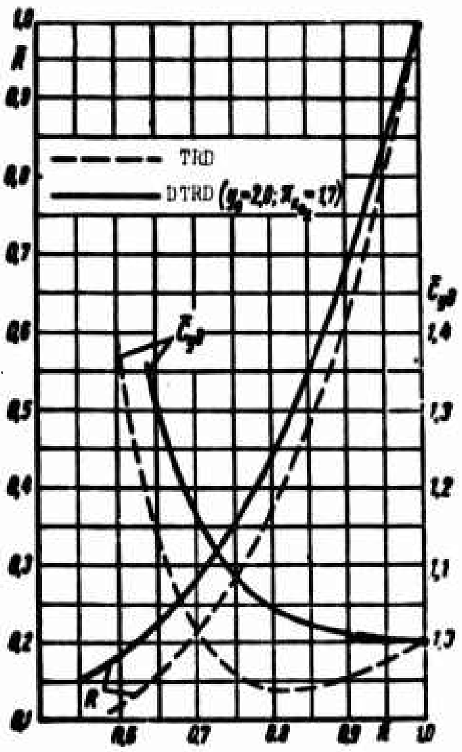


Fig. 4.14. Throttle characteristics of ducted-fan and turbojet ($H=0$, $c_0=0$, $T_3^*=1200°$ K, $\pi^*_{\text{к}_{II(0)}}=1{,}7$, $\pi^*_{\text{к}_{I(0)}}=8$, $\eta^*_{\text{к}}=0{,}85$, $\eta^*_{\text{т}}=0{,}90$).

when $\bar{n}=0{,}65$ for the ducted-fan engine $\bar{R}=0{,}22$ and $\bar{C}_{\text{уд}}=1{,}3$; accordingly for the turbojet engine $R=0{,}16$ and $\bar{C}_{\text{уд}}=1{,}14$.

It is characteristic that the curve of specific fuel consumption of a geometrically fixed single-shaft ducted-fan engine does not have a minimum inherent to characteristics of the usual single-spool turbojet engine, as a consequence of which the maximum thrust conditions of a ducted-fan engine coincide with the conditions of its best economy. In this respect the throttle characteristics of DTRD resemble the characteristics of turboprop engines.

The continuous rise of specific fuel consumption of a geometrically fixed ducted-fan engine with lowering of revolutions is physically explained by deterioration of the effective efficiency of the cycle at relatively small "specific" gravity of losses in the DTRD with exhaust velocity. At the same time, for the turbojet engine a decrease of considerable losses of kinetic flow energy during engine throttling leads to the appearance of minimum $C_{\text{уд}}$. Continuous rise of $C_{\text{уд}}$ during throttling permits leads to the fact that even at a small degree of throttling the economy advantage of the

ducted-fan engine over the turbojet vanishes. Under conditions of rougher throttling the specific fuel consumption of the TRD turns out to be substantially less than for a DTRD (Fig. 4.15a).

Figure 4.16 shows the throttle characteristic of a single-shaft ducted-fan engine of the Rato firm A-65. From the figure it is clear that the above-noted properties of throttle characteristic of single-shaft ducted-fan engine are preserved at low compression ratios ($\pi^*_{\kappa} = 4$). With decrease of the number of revolutions the temperature drop T^*_3 is small and hardly comprises 150°; in this case the specific fuel consumption of the engine continuously rises.

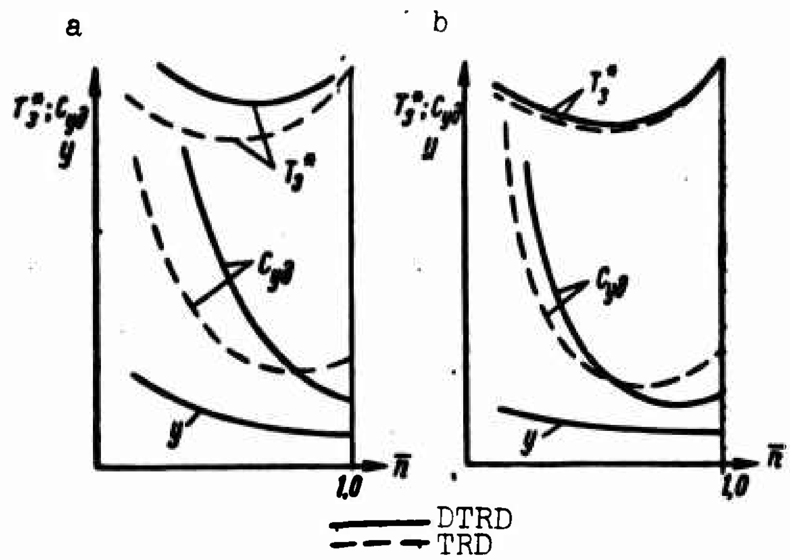


Fig. 4.15. Comparison of throttle characteristics of ducted-fan and turbojet engines: a) single-shaft engines; b) two-shaft engines.

Pickup of Single-Shaft Ducted-Fan Engine.[1]

The higher values of the gas temperature before the turbine of a ducted-fan engine as compared to turbojet in the entire range of operational conditions attest to the smaller reserve of excess power to the turbocompressor shaft

$$\Delta N = N_{\text{т}} - N_{\text{к}},$$

[1]See in this Chapter IX for more detail.

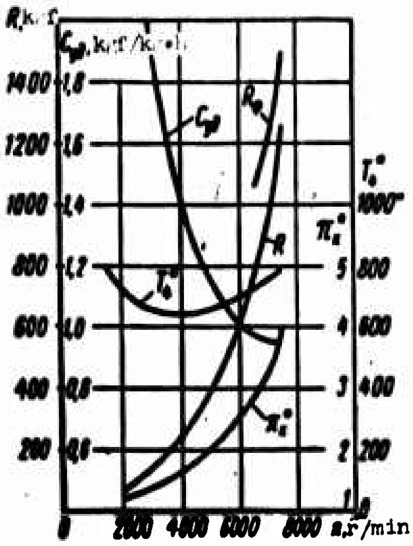


Fig. 4.16. Throttle characteristic of single-shaft Rato A-65 ducted-fan engine.

i.e., the difference between the available turbine power (determined by maximum permissible gas temperature $T_{3\text{макс}}$) and the required compressor power (determining the gas temperature before the turbine under given equilibrium conditions). This leads to increase of the engine acceleration time (arrival at performance) – impairment of pickup.

Easing of starting (including preventing an impermissible excess of gas temperature) and improvement of the pickup of a single-shaft ducted-fan engine can be attained by:

– opening the jet nozzle of the secondary duct;

– excluding the secondary duct (by covering the blades of the rotary stator (at the compressor inlet);

– opening of the jet nozzle of the primary duct;

– conversion to the two-shaft version of ducted-fan engine.

With opening of the jet nozzle of the secondary duct the power of the compressor of the secondary duct drops (since the airflow rate rises insignificantly with this, and the degree of increase

of pressure is substantially decreased). Consequently, the excess power of the turbocompressor rises. The same effect is attained by covering the blades of the stator, installed at the compressor inlet of the secondary duct (the flow rate of air through the spool is lowered because of the drop of $\pi^{*}_{\kappa II}$ and, furthermore, the effective work of the compressor is reduced).

Complete covering of the blades at the compressor inlet of the secondary duct reduces the ducted-fan engine to the operating conditions of a turbojet. However, this method does not prevent some expenditure of power for friction and heating of air circulating in the interlobe spaces during rotation of the rotor. Furthermore, with the compressor excluded the turbine passes into partial load condition of work, characterized by additional losses.

With opening of the jet nozzle of the primary duct the available power of the turbine increases due to increase of the pressure drop in it.

Finally, the transition to a two-shaft scheme of the ducted-fan engine reduces the regularity of change of T^{*}_{3} depending on the number of revolutions to the case of a two-shaft turbojet engine, and consequently, engine pickup improves.

Throttle Characteristics of Two-Shaft Ducted-Fan Engines

The tendency to eliminate operational deficiencies, inherent to single-shaft ducted-fan engines, and to improve basic indices of these engines of rated operating conditions[1] lead to the appearance of two-shaft ducted-fan engines even at the early stage of their development (Conway engines).

[1]Application of the two-shaft scheme permits increasing the pressure capacity of the high-pressure compressor stages of the primary duct at the expense of bringing the peripheral velocity of its blades to a limiting value. This gives the possibility of decreasing the dimensional length and weight per horsepower.

At present most of the series and experimental ducted-fan engines are made two-shaft.

The Effect of Throttling on the Process of Gas Expansion in a Two-Shaft DTRD Turbine.

The process of gas expansion in a two-shaft DTRD turbine during engine throttling occurs just as in any multistage turbine with exhaust to the external atmosphere. With decrease of the revolutions the pressure drop in the turbine is lowered, starting from its last stage (the one located nearest the external atmosphere). This lowering of $\pi_т^*$ gradually encompasses the stages installed upstream.

Let us assume that at the rated engine operating conditions ($n = n_1$) the gas exit velocities from the jet nozzle of the primary duct, nozzle boxes of high- and low-pressure turbines reach critical values, i.e.,

$$q(\lambda_5^1) = q(\lambda_{CA}^1)_{BД} = q(\lambda_{CA}^1)_{HД} = 1.$$

With engine throttling in the supercritical region of exhaust from the jet nozzle $(\pi_{p.c} > \pi_{кр} = 1{,}85)$ a lowering of the engine expansion ratio occurs at first due to decrease of the pressure on the nozzle section. Pressure drops in high-pressure and low-pressure turbines remain constant (Fig. 4.17).

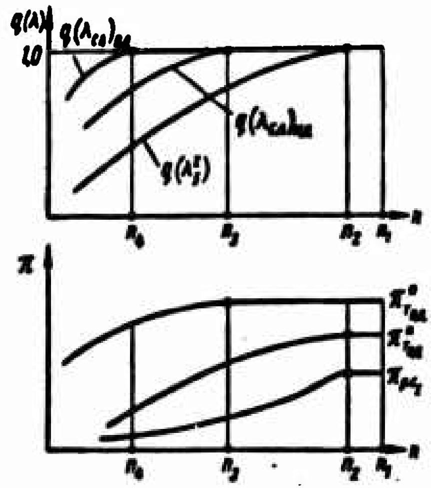


Fig. 4.17. Redistribution of pressure drop between high-pressure and low-pressure turbines during throttling of ducted-fan engine.

At a certain value of the number of revolutions $n_2 < n_1$ in the jet nozzle there are established subcritical exhaust conditions, i.e., $q(\lambda_{\text{Д}}^{\text{I}}) < 1$. Starting from this moment, the jet nozzle inlet pressure drops. The expansion ratio in the low-pressure turbine is simultaneously lowered. Inasmuch as exhaust conditions from the primary nozzle box of the low-pressure turbine still continue to be kept critical, the high-pressure turbine remains "locked" with respect to pressure drop, i.e., $\pi^*_{\text{т}_{\text{ВД}}}$ = const. With further lowering of revolutions to n_3 the exit velocity from the primary nozzle box of the low-pressure turbine becomes subcritical. From this moment the pressure differential in the high-pressure turbine starts to drop. Now the decreasing total pressure drop is redistributed between the high-pressure and low-pressure turbines. As $\pi^*_{\text{т}_{\text{НД}}}$ approaches 1 the drop of $\pi^*_{\text{т}_{\text{ВД}}}$ is accelerated.

Figure 4.18 shows the experimental dependence[1] of the change of $q(\lambda^{\text{I}}_{\text{СА}})$ on $\pi^*_{\text{т}}$.

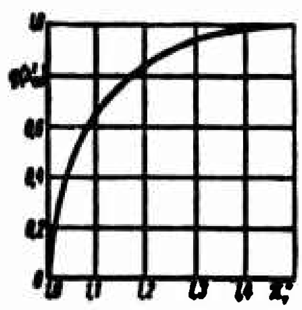


Fig. 4.18. For determination of operating conditions of high-pressure turbine.

Since

$$\pi_{\text{т}_{\text{ВД}}}^{*\frac{n+1}{2n}} = \frac{f^{\text{I}}_{\text{СА(НД)}}\, q(\lambda^{\text{I}}_{\text{СА}})_{\text{НД}}}{f^{\text{I}}_{\text{СА(ВД)}}\, q(\lambda^{\text{I}}_{\text{СА}})_{\text{ВД}}} \sim q(\lambda^{\text{I}}_{\text{СА}})_{\text{НД}},$$

this graph shows the qualitative connection of the drop of $\pi^*_{\text{т}_{\text{ВД}}}$ as a function of $\pi^*_{\text{т}_{\text{НД}}}$.

[1]According to Mayer [20].

Change of the Gas Temperature Before the Turbine of a Two-Shaft Ducted-Fan Engine.

The regularity of change of temperature T_3^* depending on the number of revolutions of a two-shaft ducted-fan engine is completely determined by the balance of works of high-pressure turbocompressor

$$L_{\text{твд}}=L_{\text{квд}},$$

whence

$$T_3^*=\frac{L_{\text{квд}}}{\frac{c_{p_\text{г}}}{A}\,\varepsilon_{\text{твд}}^*\,\eta_{\text{твд}}^*}. \qquad (4.23)$$

Since the primary nozzle box of the low-pressure turbine in the initial throttling stage works under a critical pressure drop, then $\pi_{\text{твд}}^*$ = const and $T_3^*\sim L_{\text{квд}}$. In this range of revolutions there sets in lowering of T_3^*, which is determined by decrease of the work of high-pressure compressor depending on the number of revolutions. With further engine throttling the pressure drop on the high-pressure turbine is reduced [when $q(\lambda_{\text{CA}})_{\text{нд}}<1$], as a consequence of which the drop of T_3^* is delayed. As the revolutions decrease the gas temperature before the turbine reaches minimum, and then starts to rise.

Regularity of change of T_3^* in a wide range of revolutions of a two-shaft ducted-fan engine turns out to be the same as for a two-shaft turbojet engine.

"Slip" of High- and Low-Pressure Turbocompressors.

With reductions of the gas temperature before the high-pressure turbine (T_3^*) the gas temperature in front of the low-pressure turbine ($T_{4\text{нд}}^*$), is lowered and the bypass ratio increases. In view of this the balance of works is disturbed on the low-pressure turbocompressor

$$(1+y)L_{\text{кнд}}>L_{\text{тнд}}.$$

As a result the revolutions of the low-pressure turbocompressor drop more intensively than the high-pressure. The drop of revolutions of the low-pressure turbocompressor is intensified when the pressure drop in the jet nozzle of the primary duct becomes subcritical and a drop of the pressure differential sets in at the low-pressure turbine of the engine (Fig. 4.19).

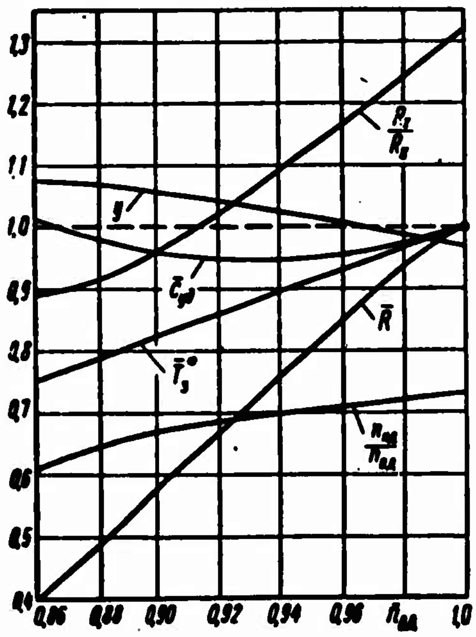


Fig. 4.19. Throttle characteristic of a two-shaft ducted-fan engine ($H=0$, $M_0=0$, $y_p=1$, $T^*_{3(p)}=1300°$ K, $C_{T\pi_p}=0.7\ \frac{кГ}{кГ\cdot ч}$, $\pi^*_{кI(p)}=13$, $\pi^*_{кII(p)}=2.7$).

Thus, during engine throttling the relationship of revolutions $\bar{n}=\frac{n_{НД}}{n_{ВД}}$ is continuously lowered. We call this phenomenon turbocompressor "slip".

<u>Change of the Engine Bypass Ratio While Throttling.</u>

With decrease of revolutions the bypass ratio of the two-shaft ducted-fan engine is increased. Increase of y is caused by the difference of compression ratios of compressors in the spools ($\pi^*_{кII} \ll \pi^*_{кI}$) and, consequently, by faster drop of the flow rate of air through the primary duct as compared to the secondary. However, an advanced drop of low-pressure compressor revolutions accelerates the drop of $\pi^*_{кII}$ and delays the rise of y.

This substantially improves the throttle characteristics of two-shaft ducted-fan engines as compared to single-shaft (see Fig. 4.15b).

Redistribution of DTRD Thrust Between Ducts During Throttling.

On Fig. 4.20 there is shown change of the gas exit velocity from the jet nozzles of a two-shaft ducted-fan engine during engine throttling.

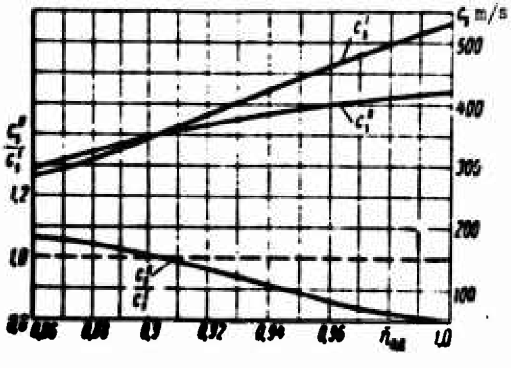


Fig. 4.20. Change of the gas exit velocity from jet nozzles of a ducted-fan engine during engine throttling $(H=0, M_0=0, y_p=1, T^*_{3(p)}=1300°K, \pi^*_{K\,I(p)}=13, \pi^*_{K\,II(p)}=2.7)$.

It is characteristic that with decrease of engine revolutions the ratio of exit velocities from spools $\frac{c_5^{II}}{c_5^{I}}$ continuously increases, reaching 1 when n = 0.9. With further decrease of revolutions the ratio $\frac{c_5^{II}}{c_5^{I}}$ continues to rise. Thus, the ratio of exit velocities is deflected from optimum value $\left(\frac{c_5^{II}}{c_5^{I}} = \eta_{II} = 0.8\right.$ at maximum conditions), which attests to impairment of the distribution of energy between ducts and leads to relative increase of the specific fuel consumption as compared to a turbojet engine. At the same time the relative increase of specific thrust of the secondary duct, and also the growth of parameter y lead to considerable redistribution of thrusts between the ducts of a ducted-fan engine. Thus, if when $\bar{n}_{вд}=1$ (rated conditions) $\frac{R_I}{R_{II}}$ = 1.32, then when $\bar{n}_{вд}$ = 0.86 we already have $\frac{R_I}{R_{II}}$ = 0.89.

Change of Thrust Depending on the Number of Revolutions.

On Fig. 4.21 there is presented the throttle characteristic of a two-shaft ducted-fan engine, constructed depending on the number of revolutions of the low-pressure turbocompressor.

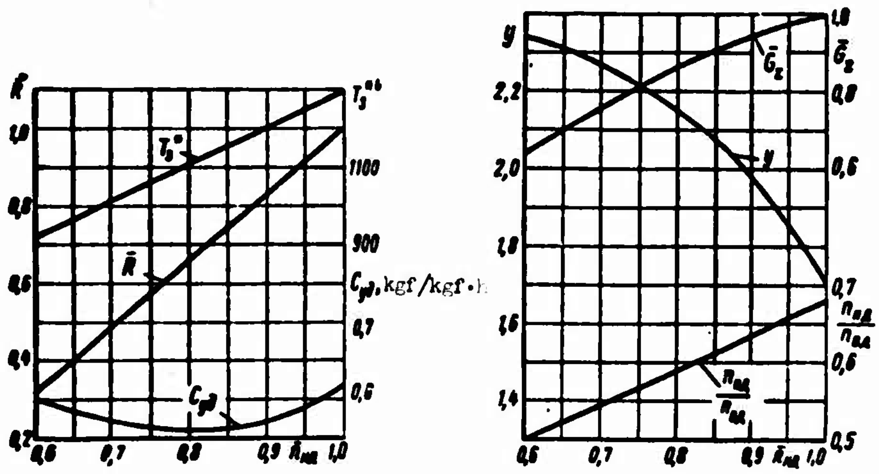


Fig. 4.21. Throttle characteristic of a two-shaft ducted-fan engine ($M_0=0$, $H=0$, $\pi^*_{K1(p)}=12.5$, $T^*_{3(p)}=1300°$ K).

Figure 4.22 shows the change of thrust of two-shaft ducted-fan engines, measured on the stand, having different bypass ratios

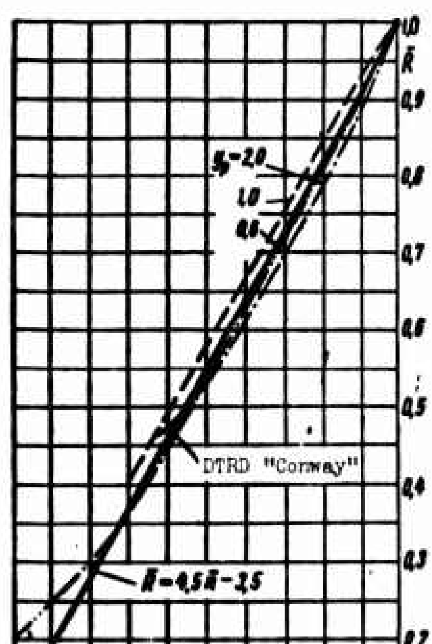


Fig. 4.22. Change of DTRD thrust depending on the number of revolutions ($H = 0$, $M_0 = 0$).

(y = 2; 1; 0.6). The engines have a high compression ratio of the primary duct ($\pi^*_{\kappa 1(0)}$ = 13-15) and high-temperature turbine (T_3^* = 1300°K). From the figure it is clear that the curves have the form of a straight line and hardly differ from each other. In the region of high revolutions they can be approximated by equation

$$\bar{R} = 4{,}5\,\bar{n} - 3{,}5.$$

Comparison of the Throttle Characteristics of Two-Shaft Ducted-Fan and Turbojet Engines

On Fig. 4.23 there are given the comparative throttle characteristics of two-shaft ducted-fan and turbojet engines under calculated conditions at

$$T_3^* = 1300° = idem;\quad \pi^*_{\kappa_1} = 12{,}5 = idem;\quad y = 1;\quad G_1 = idem.$$

From the figure it is clear that the transition to a two-shaft scheme permits considerably expanding the range of working revolutions, in which advantages of the DTRD over the single-spool turbojet engine are provided with respect to specific fuel consumption.

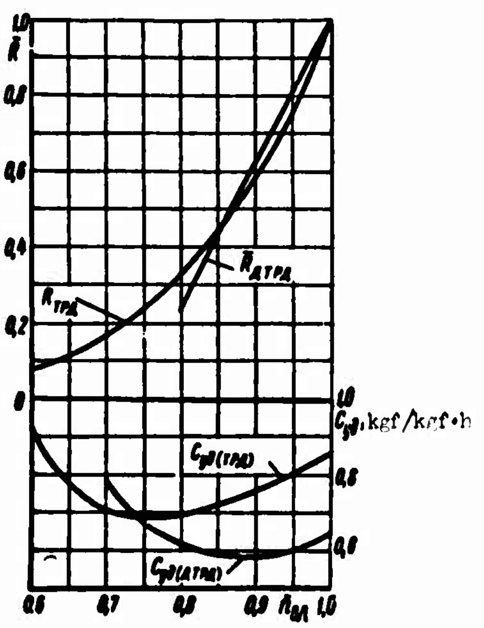


Fig. 4.23. Comparison of throttle characteristics of two-shaft DTRD and turbojet engines ($H=0$, $M_0=0$, $y=1$, $T_3^*=1300°$ K, $\pi_\kappa^*=12{,}5$).

However, a more intense drop of revolutions of the low-pressure turbocompressor for the ducted-fan engine and faster decrease of total compression ratio, connected with this, lead to the fact that at a certain number of revolutions ($\bar{n}_{вд} \approx 0{,}75$) the specific fuel consumptions of engines are equalized. With further throttling the specific fuel consumption of a ducted-fan engine becomes greater than for a turbojet engine. Impairment of the operating economy of the ducted-fan engine under conditions of rough throttling contributes to a sharp drop efficiency of the multistage turbine.

In conclusion let us note that two-shaft ducted-fan engines with a forward location of the fan in practice are not inferior in their operational qualities to the best single-spool turbojet engines.

Throttle Characteristic of Ducted-Fan Engine with Aft Location of Fan

We will compare the throttle characteristics of ducted-fan engines with aft location of fan and the initial TRD for this engine. The presence of a free turbocompressor does not affect the gas parameters past the high-pressure turbine during engine throttling (with accuracy up to the efficiency of the turbine).

With decrease of fuel feed to the combustion chamber of the engine the consumption is lowered; the pressure and temperature of gas past the turbine are also lowered, i.e., the power of the turbine of the free turbocompressor drops, while ratio $\frac{n_{нд}}{n_{вд}}$ is continuously lowered during throttling. The engine bypass ratio at first increases and then starts to drop intensively. In some cases engine parameter y is continuously lowered during throttling.

On Fig. 4.24 there are presented comparative throttle characteristics of a ducted-fan engine (with free turbocompressor) and initial turbojet engine, constructed in relative magnitudes for values of parameters of the working process under rated conditions: $T_3^* = 1500^{\circ}K$, $\pi_к^* = 15$, y = 2. From the figure it is clear that the advantages of the ducted-fan engine in thrust and economy over the initial turbojet engine is preserved in practically the entire range

of working revolutions.

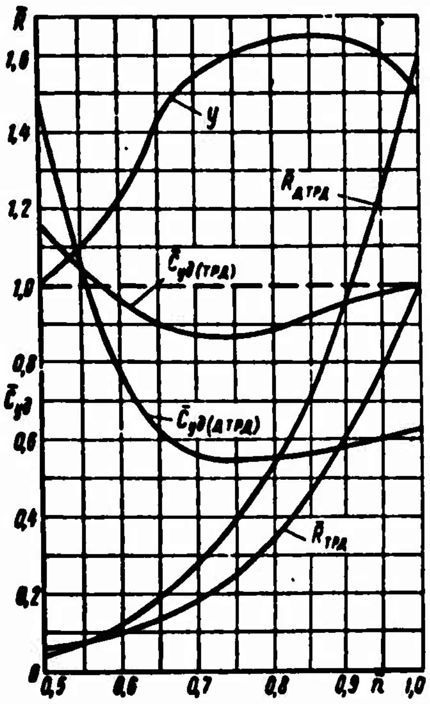


Fig. 4.24. Throttle characteristics of ducted-fan engine with aft location of fan ($H = o$, $M_0 = 0$, $y_0 = 2$, $T_3^* = 1500^oK$, $\pi_{\kappa I(0)} = 15$).

Theoretically, with the absence of losses for the free turbocompressor ($\eta_\text{т}^* \eta_\text{к}^* = 1$) in the entire range of working revolutions the thrust of a ducted-fan engine should be greater, and specific fuel consumption should be less than for a turbojet engine. In reality the presence of hydraulic losses in the gas-air duct of the free turbocompressor permits leads to the fact that at a certain number of revolutions (in the given example $\bar{n} = 0{,}55$) the thrust and economy of both engines are equalized.

§ 3. High-Speed Characteristics of DTRD

Approximation Methods of Calculation of the High-Speed Characteristics of Ducted-Fan Engines

Exact calculation of the high-speed characteristics of ducted-fan engines is a very complicated problem. Besides the many calculation operations, it requires the use of the characteristics of separate engine elements and primarily — characteristics of the compressors in each of the spools.

By knowing the course of lines of operating conditions on the compressor characteristics, for the prescribed number of physical revolutions (n) of the engine in accordance with change of the flight altitude and speed (and consequently, T^*_H) it is possible to calculate the change of the number of given revolutions ($n_{пр}$) and to determine by the values of $L_к$ and $\eta^*_к$ the compressor characteristic. Further calculation of ducted-fan engine parameters is not difficult.

Results of such calculations are extremely "individualized"; they depend not only on parameters of the working process under calculated operating conditions of the engine, but to a considerable degree on the peculiarities of compressor characteristics and the program of control. For fast completion of similar labor-consuming calculations electronic computers have been used to an increasingly greater degree during the last few years.

Besides exact calculations of high-speed characteristics of ducted-fan engines, in the practice of engine manufacture there appears a need for the development and use of simplified methods of calculation of ducted-fan engine characteristics. They are expedient, for example, for variant "rough calculations" when designing an engine in the experimental design office, and for students to obtain approximate engine characteristics in course and graduate designing.

Generalized Characteristics of Axial-Flow Compressors and Turbines.

Numerous investigations of compressor characteristics permitted revealing generalized regularities of the change of parameters $L_к$ and $\eta^*_к$ as functions of $n_{пр}$ (or T^*_H) at programs of control n = const.

With decrease of $n_{пр}$ (i.e., rise of T^*_H) the angles of attack on the rotor blades of the first stages of the compressor are increased and are decreased on the last-stage blades. With increase of $\pi^*_к$ the increase of head of the first stages becomes determining and causes increase of work of the entire compressor. Conversely, at small values of $\pi^*_к$ the total work of the compressor is lowered. At

$\pi^*_{\kappa_0} = 6$ the change of $n_{пр}$ does not practically affect the compressor work $(L_к \approx \text{const})$.

Treatment of the experimental characteristics of many compressors permitted the Soviet scientist R. M. Fedorov to find semiempirical dependencies of the effect of the computed value of $\pi^*_{\kappa_0}$ on the regularity of change of $L_к = f(n_{пр})$ when n = const and f_5 = const. Refined analogous dependencies, obtained by treatment of many characteristics of compressors of various series, are given on Figs. 4.25 and 4.26 according to N. D. Tikhonov.

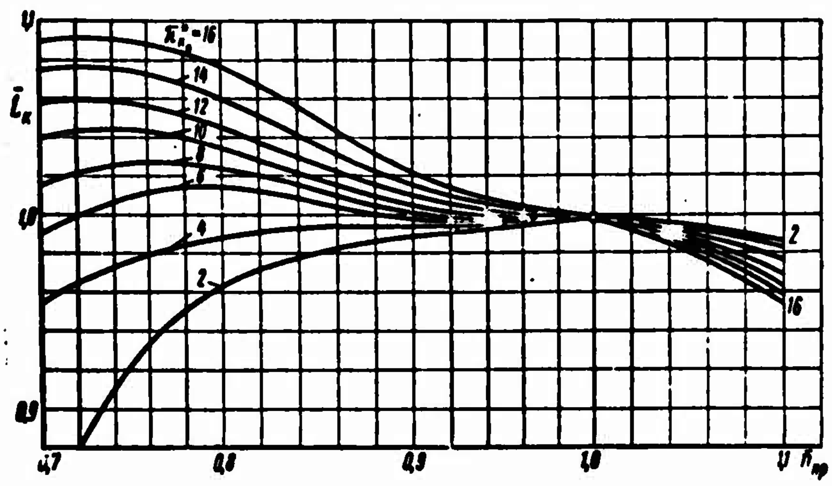


Fig. 4.25. The effect of $\pi^*_{\kappa_0}$ on relationship $\bar{L}_к = f(\bar{n}_{пр})$ when n = const.

It is necessary to note that with increase of T^*_H the compressor efficiency is changed; it is changed more intensely, the greater the value of $\pi^*_к$ at initial (rated) conditions. At large values of $\pi^*_к$ there sets in an intense drop of $\eta^*_к$. It is explained by deflection of the line of operating conditions from the zone of maximum compressor efficiency.

The dependencies shown in Figs. 4.25 and 4.26 can be considered as certain generalized compressor characteristics. They can be used for calculation of the high-speed characteristics of ducted-fan engines in cases when there are no real characteristics of the compressors of primary and secondary ducts.

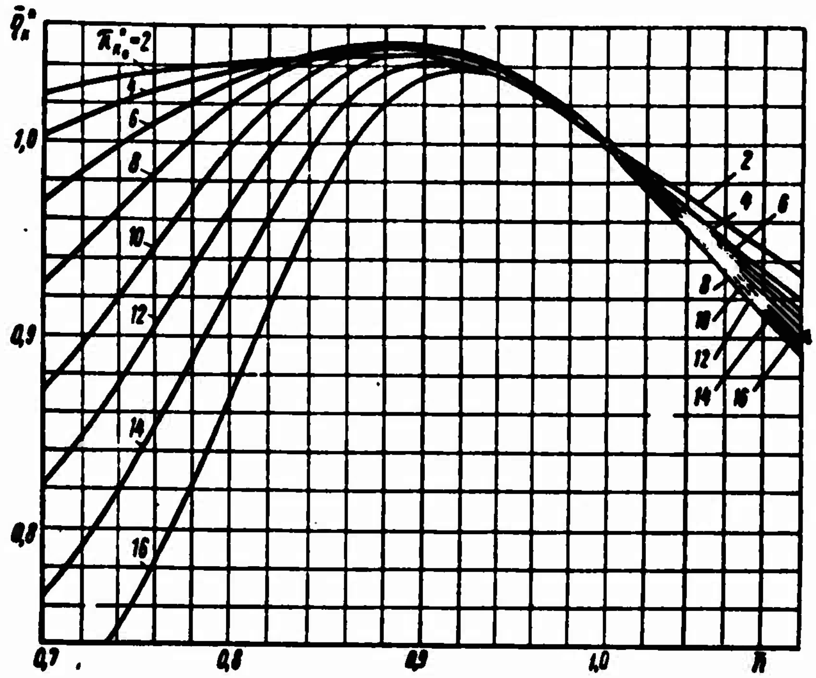


Fig. 4.26. Effect of $\pi_{\text{к}_0}^*$ on relationship $\bar{\eta}_\text{к}^* = f(\bar{n}_{\text{пр}})$ when n = const.

As generalized turbine characteristics there are widely used dependencies

$$\frac{G_\text{г}}{p_3^*}\sqrt{T_3^*}=\text{const},\ L_\text{т}=\text{const и } \eta_\text{т}^*=\text{const}.$$

These equalities are observed precisely in cases when pressure differential on the turbine remains constant ($\pi_\text{т}^*$ = const) and the const given number of revolutions $n_{\text{т пр}}$ = const. In this case the turbine works under similar conditions. Usually the first two dependencies are fulfilled in flight. Regarding the turbine efficiency, it is changed, since $n_{\text{пр}}\neq\text{const}$ or $\frac{u}{c_1}\neq\text{const}$. However, the change of $\eta_\text{т}^*$ turns out to be insignificant.

Approximate Relations for Basic Parameters of Compressors.

In some cases in calculation practice simplified relations are used when evaluating compressor operation in flight in the form

of $L_к$ = const when n = const and $\eta_к^*$ = const.[1]

Constancy of compressor work is assured with change of T_H^* when the rated compression ratio of the compressor of the basic spool of turbojet engine (ducted-fan engine) is approximately equal to 6 (Fig. 4.25). In the secondary duct of a ducted-fan engine this condition is observed rather exactly along the line of maximum efficiencies of the compressor.

Investigations show that in the subsonic flight speed range the altitude and high-speed characteristics of a nonreheated DTRD with fixed geometry can be calculated by using the above-described assumptions. In this case the errors are equal to 1-2%.

However, propagation of these relationships to a wide range of supersonic flight speeds can lead to gross errors (up to 10-12% with respect to parameters $L_к$ and $\eta_к^*$). This especially concerns the assumption of constancy of efficiency.

High-Speed Characteristics of Single-Shaft Ducted-Fan Engines

Nonreheated Ducted-Fan Engine

Let us examine the high-speed characteristic of a single-shaft nonreheated ducted-fan engine with program of control for maximum thrust: n = const, T_3^* = const.

Let us assume that at rated (stand) operating conditions (M_0 = 0; H = 0) the parameters of the engine working process are equal

$$T_3^*=1200°\,K;\quad \pi_{к_{I(0)}}^*=15;\quad \pi_{к_{II(0)}}^*=2,15;\quad y=1;$$
$$\eta_т^*=0,90;\quad \eta_к^*=0,85.$$

[1]Relationships $L_к$ = const and $\eta_к^*$ = const can be considered as characteristics compressor with a sufficiently refined control system.

Distribution of energy between spools on the stand provides equal pressure drop in the jet nozzles of the spools, i.e., $\pi_{p.c_I} = \pi_{p.c_{II}}$.

Constant turbocompressor revolutions are maintained with the aid of a centrifugal speed regulator, interlinked with the automatic fuel feed control. Constancy of gas temperature before the turbine is attained by regulating the jet nozzle throat of the primary duct (f_5^I = var).

At first let us examine the high-speed characteristics of ducted-fan engines, obtained as a result of approximate calculation, without the use of characteristics of compressors, turbines and intakes.

The basic assumptions, usually taken in approximate calculations of characteristics, include:

1) constancy of work of compressors (for both ducts):

$$L_к = \text{const (when } n = \text{const)};$$

2) constancy of particular efficiencies and loss factors:

$$\eta_к^* = \text{const},\ \eta_т^* = \text{const},\ \sigma_{к.c}^* = \text{const},\ \varphi_{р.c} = \text{const},\ \xi_{к.c} = \text{const};$$

3) total expansion of gas in the jet nozzles of both ducts:

$$p_5^I = p_5^{II} = p_H.$$

Change of Specific Thrust of Ducted-Fan Engines.

On a stand the gas exit velocity from the primary duct is greater than from the secondary. This is explained by the fact that at equal pressure drops in the jet nozzles the gas temperature past the turbine is considerably higher than the air temperature at the secondary duct compressor exit. From this it follows that the specific thrusts of the ducts are also different:

$$R_{y\text{д}_{II}} < R_{y\text{д}_I}.$$

With increase of the M_0 number of flight the specific thrusts of the ducts drop continuously. However, the specific thrust of the primary duct decreases more intensively, since the pressure drop in the jet nozzle of this duct grows considerably slower than in the secondary. Opening of the jet nozzle of the primary duct for maintaining T_3^* = const intensifies this tendency still more as the flight speed increases (Fig. 4.27). The specific thrust of ducted-fan engines decreases analogously, though considerably faster than for the initial turbojet engine. If for a turbojet engine the specific thrust becomes zero at number $M_0 \approx 2{,}8$, for a ducted-fan engine it becomes zero at number $M_0 \approx 2{,}4$ (Fig. 4.28).

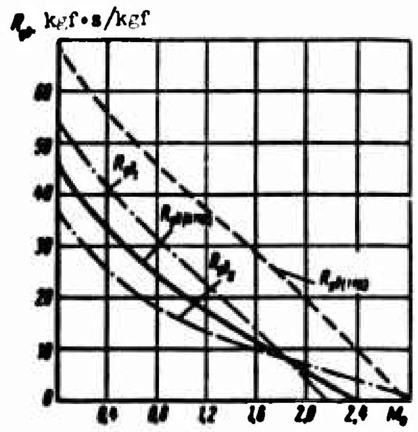


Fig. 4.27. Change of specific thrusts of DTRD ducts depending on the M number of flight $(H=0,\ T_3^*=1200°\text{ K},\ y=1,\ \pi^*_{\text{к}_{I(0)}}=15,\ \pi^*_{\text{к}_{II(0)}}=2{,}15)$.

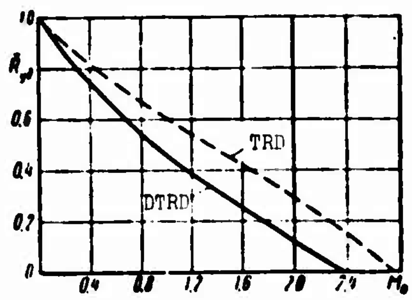


Fig. 4.28. Comparative change of specific thrusts of ducted-fan and turbojet engines depending on the M number of flight $(H=0,\ y=1,\ T_3^*=1200°\text{ K},\ \pi^*_{\text{к}_{I(0)}}=15,$ $\pi_{\text{в}II(0)}=2{,}15,\ \pi_{p.c_I}=\pi_{p.c_{II}})$.

<u>Change of Bypass Ratio.</u>

With increase of flight speed the flow rate of air per second through the secondary duct rises; it rises much faster than through

the primary duct ($\pi^*_{\kappa_{II}} \ll \pi^*_{\kappa_I}$). This leads to the fact that the engine bypass ratio grows intensely. If at number $M_0 = 0$ the value of $y = 1$, then when $M_0 = 2.5$ the value of $y = 1.68$ (Fig. 4.29).

The rapid increase of the bypass ratio of the engine as a function of M_0 number leads to increase of the required work of the turbine, equal to

$$L_{\tau} = L_{\kappa_I} + yL_{\kappa_{II}}.$$

For preservation of T^*_3 = const it is necessary to increase π^*_{τ}, with rise of the M_0 number, and consequently, to open the jet nozzle of the primary duct.

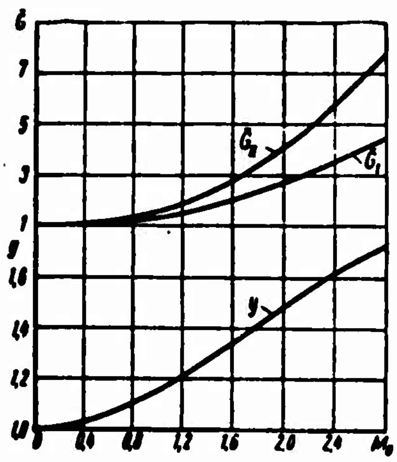


Fig. 4.29. Change of the airflow rate in DTRD ducts depending on the M number of flight ($H_0 = 0$, $y = 1$, $T^*_3 = 1200^{\circ}K$, $\pi^*_{\kappa_{I(0)}} = 15$, $\pi^*_{\kappa_{II(0)}} = 2.15$, $\pi_{p.c_I} = \pi_{p.c_{II}}$).

Change of Total Thrust of Ducted-Fan Engines.

Peculiarities of curves of specific thrusts and airflow rates in ducts are determined by the regularity of change of total thrusts R_I, R_{II}, and R. In accordance with Fig. 4.30, thrust of the secondary duct drops considerably slower than the primary. If at subsonic flight speeds $R_I > R_{II}$, then at high supersonic flight speeds $R_{II} > R_I$ (Fig. 4.30).

Total DTRD thrust is also continuously lowered with increase of the flight speed, also the rate of its drop is incomparably faster than for the initial turbojet engine (Fig. 4.31).

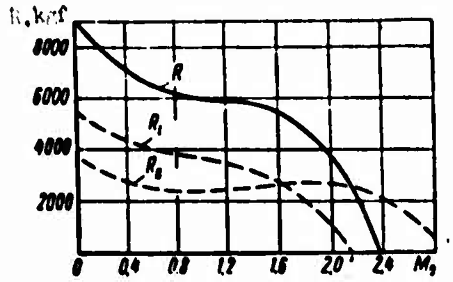


Fig. 4.30. Change of thrusts of DTRD ductd depending on the M number of flight $(H=0,\ y=1,\ T_3^* = 1200°\ K,\ \pi^*_{k_{I(0)}} = 15,\ \pi^*_{k_{II(0)}} = 2.15,\ \pi_{p.c_I} = \pi_{p.c_{II}})$.

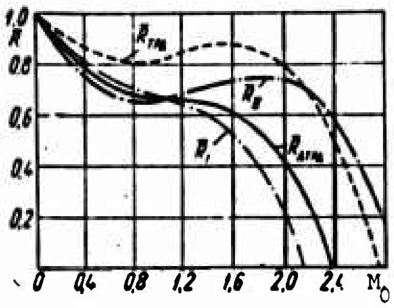


Fig. 4.31. Change of thrusts of ducted-fan and turbojet engines depending on the M number of flight $(H=0,\ y=1,\ T_3^* = 1200°\ K,\ \pi^*_{k_{I(0)}} = 15,\ \pi^*_{k_{II(0)}} = 2.15,\ \pi_{p.c_I} = \pi_{p.c_{II}})$.

It is necessary to note that the greater the engine bypass ratio, the more sharply thrust drops with increase of the flight speed and the smaller the M_0 number on which engine thrust approaches zero (Fig. 4.32).

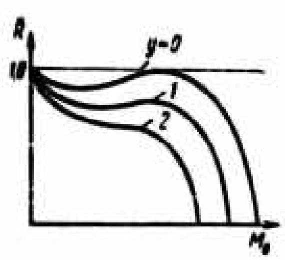


Fig. 4.32. Effect of the bypass ratio on thrust characteristics of ducted-fan engines depending on the M number of flight.

Change of DTRD Efficiency Depending on the Flight Speed.

Let us compare the change of efficiency (effective, thrust and overall) of single-spool TRD and DTRD depending on the flight speed (Fig. 4.33).

Transmission of mechanical energy through the turbine and compressor of the secondary duct decreases exit the gas velocity from the primary duct and increases the gas exit velocity from the secondary duct. However, with the applied methods of distribution

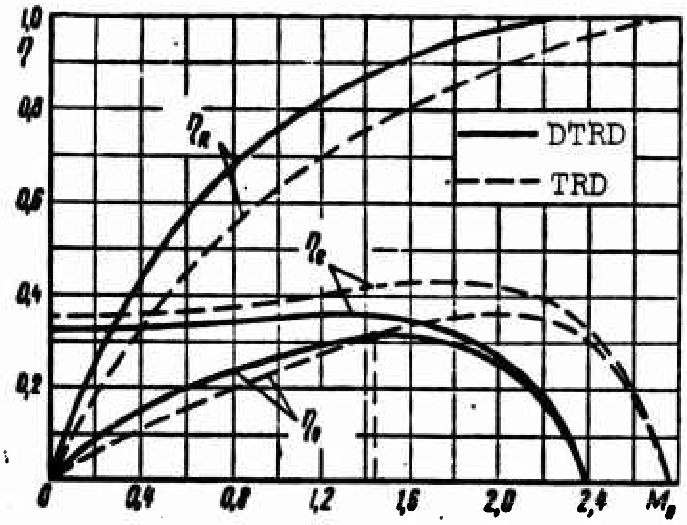


Fig. 4.33. Change of DTRD efficiency depending on the M number of flight ($H=0$, $y=1$, $T_3^* = 1200°$ K, $\pi^*_{K_{I(0)}} = 15$, $\pi^*_{K_{II(0)}} = 2.15$, $\pi_{p.c_I} = \pi_{p.c_{II}}$).

of energy between ducts the gas exit velocity from the primary duct is always smaller than from the primary (by 30-40%). In any case, the numerical values of c_5^I and c_5^{II} are considerably less than the exit velocities from the jet nozzle of the initial turbojet engine. Therefore, the thrust efficiency of a ducted-fan engine at all flight speeds (especially at subsonic) is substantially greater than for a single-spool turbojet engine.

In connection with the fact that the transmission of mechanical energy into the secondary engine duct occurs with losses, the effective efficiency of a ducted-fan engine is less than for the usual turbojet; this decrease becomes especially noticeable at high supersonic flight speeds. At these flight speeds the transmission of mechanical energy is generally hardly effective, since it considerably decreases the thrust of the primary duct and immaterially increases the thrust in the secondary duct.

As a result the overall efficiency of a ducted-fan engine at the examined working process parameters somewhat exceeds the TRD efficiency in the range of M_0 numbers from 0 to 1.4. At high supersonic flight speeds ($M_0 > 1.4$) the overall TRD efficiency is even higher than for the ducted-fan engine.

Change of Specific Fuel Consumption.

On a stand the specific fuel consumption of a ducted-fan engine at the above-mentioned working process parameters is equal to $C_{уд} = 0.60$; it is considerably less than for the initial turbojet engine, for which $C_{уд} = 0.79$.

With increase of M_0 number of flight the specific fuel consumption of ducted-fan and turbojet engines continuously rises, since the useful work of each kilogram of thrust is increased, thus, the energy expended in the form of fuel consumption for this kilogram of thrust per hour increases. However, the rate of increase of specific fuel consumption of a ducted-fan engine depending on the flight speed is considerably greater than for a turbojet engine (Fig. 4.34). Thus, with increase of M_0 number the discontinuity between curves $C_{уд}$ for ducted-fan and turbojet engines is increasing reduced and at number $M_0 \approx 1.4$ the engine economies are equalized. With further increase of M_0 number the specific fuel consumption of the ducted-fan engine already exceeds the TRD.

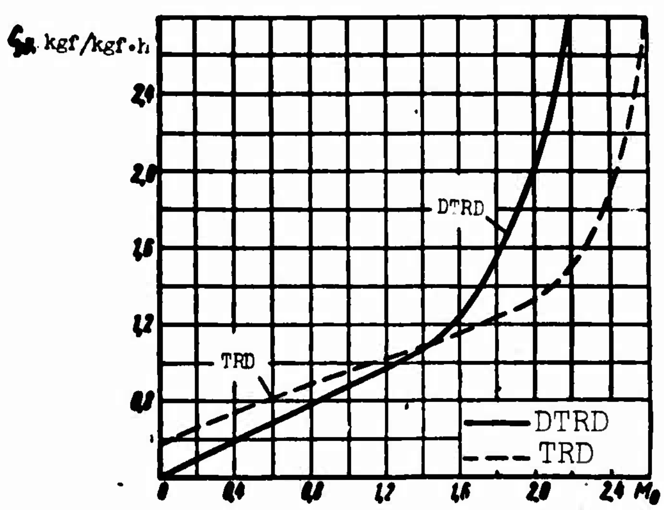


Fig. 4.34. Change of specific fuel consumption of a ducted-fan engine depending on the M number of flight ($H = 0$, $y = 1$, $T_3^* = 1200^\circ K$, $\pi^*_{к_{I(0)}} = 15$, $\pi^*_{к_{II(0)}} = 2.15$, $\pi_{р.с_I} = \pi_{р.с_{II}}$).

The relationship between specific fuel consumptions of comparable engines at the same flight speed is equal to

$$\frac{C_{уд\,(ДТРД)}}{C_{уд\,(ТРД)}} = \frac{\tau_{0\,(ТРД)}}{\tau_{0\,(ДТРД)}}$$

or

$$\bar{C}_{y\text{д}} \sim \frac{1}{\bar{\eta}_0}.$$

On Fig. 4.35 there is shown the effect of the bypass ratio on the change of $C_{y\text{д}}$ of a ducted-fan engine depending on the flight speed. The higher the value of y, the lower the stand value of $C_{y\text{д}}$ and the lower the M_0 number of flight, at which the advantage in economy of a ducted-fan engine above an initial turbojet engine vanishes.

It is characteristic that if the efficiency of the secondary duct is equal to 1, then in the entire range of M_0 numbers of flight the ducted-fan engine would be more economical than the TRD (Fig. 4.36).

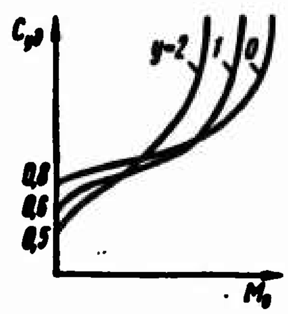


Fig. 4.35. Effect of the bypass ratio on the change of specific fuel consumption of a ducted-fan engine depending on the M number of flight.

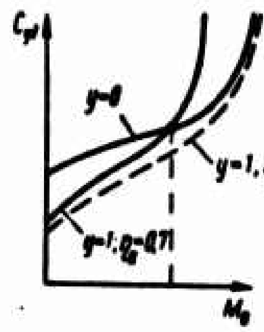


Fig. 4.36. Effect of efficiency of the secondary duct on the change of specific fuel consumption of a ducted-fan engine depending on the M number of flight.

Combined Program of Control of a Ducted-Fan Engine $n_{\text{кВД}}$ = const and $n_{\text{кНД}}$ = const

Real DTRD limitations, appearing during flight operation, in a number of cases compel the application of a program of control, being a combination of programs $n_{\text{кНД}}$ = const and $n_{\text{кВД}}$ = const (Fig. 4.37).

Let us examine the work of a ducted-fan engine with program of control $n_{\text{кНД}}$ = const and f_s = const.

Let us assume that $n_{КНД}$ – maximum number of revolutions of the low-pressure compressor, limited by the strength of this stage (region II).

The region of values of T_H^*, in which $n_{КНД}$ = const is limited on the one hand by a certain minimum value of T_H^*, at which the given number of revolutions of the low-pressure compressor reaches maximum and "blocking" of the airflow rate sets in at the compressor inlet, i.e., increase of $n_{НД(пр)}$ is ceased with rise of $q(\lambda_1)_{НД}$. Starting from this number of revolutions, it is necessary to convert to control $n_{НД(пр)}$ = const. This means that with further lowering of T_H^* the physical revolutions of thw low-pressure compressor should drop.

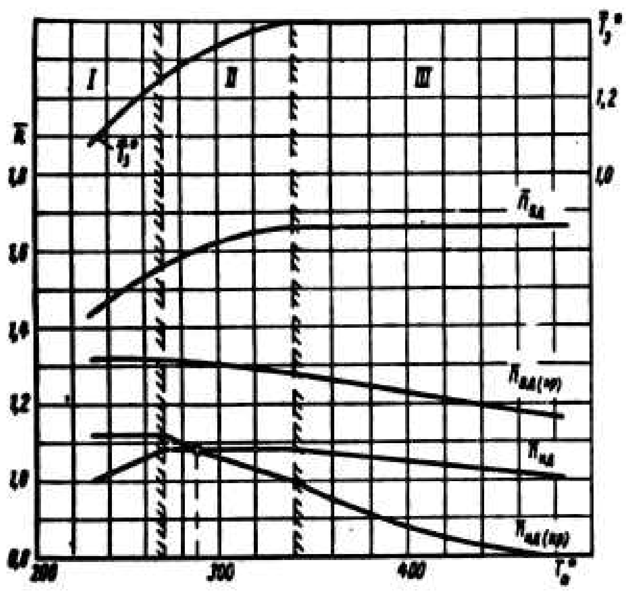


Fig. 4.37. Combined program of control of ДТРДФ".

Thus, in the region of small values of T_H^* there sets in a limitation on the maximum compressor output (region I).

In the region of high values of T_H^* limitation of the program $n_{КНД}$ = const is connected with the maximum permissible value of T_3^*. Really, as T_H^* increases $n_{КНД}$ = const is maintained by increase of fuel feed to the combustion chamber and, as a result of this, by increase of T_3^* and the number of revolutions of the high-pressure compressor. It is obvious that at a certain limiting value of T_H^*

the value of T_3^* reaches the permissible limit.

In this case for preservation of $T^*_{3(\text{макс})}$ = const it is necessary to change to control $n_{\text{КВД}}$ = const. Now $n_{\text{КНД}}$ will even be decreased (region III).

Figure 4.37 shows change of the performance parameters of DTRD for the examined case of the combined program of control.

From the figure it is clear that to section $n_{\text{КНД}}$ = const there correspond relatively small changes of $n_{\text{ВД(пр)}}$ and $n_{\text{НД(пр)}}$, and also a small shift to performance point on characteristics of the low- and high-pressure compressors; on section $n_{\text{КВД}}$ = const the given numbers of revolutions of low- and high-pressure compressors, and also displacement of the performance point of compressors sharply increase.

The Effect of Boosting on the High-Speed Characteristic of Ducted-Fan Engines

At large values of $\pi^*_{\text{к}_{\text{I}}}$ a continuous drop of thrust and intense increase of the specific fuel consumption depending on the M_0 number of flight are inherent to an unboosted ducted-fan engine.

Additional combustion of fuel in the secondary duct at high values of T^*_{ϕ} sharply increases the specific thrust on the stand and delays its drop in flight. Thus, DTRDF^{II} thrust does not have a specific dip at low flight speeds and grows very intensively in the supersonic region; only at M_0 numbers of flight, at which in the primary duct there is formed negative thrust, does a sharp drop of total engine thrust set in (Fig. 4.38).

The introduction of boosting in the secondary duct essentially increases the specific fuel consumption on the stand. With rise of M_0 numbers of flight the specific fuel consumption slowly increases.[1] At supersonic flight speeds it becomes lower than

[1]On separate sections of the characteristic there can even set in lowering of $C_{\text{уд}}$.

for the initial turbojet engine (see Fig. 4.38).

Figure 4.39 contains the characteristic of a two-shaft DTRDF^{II} with combined program of control (see Fig. 4.37). Growth of parameters T_3^* and T_Φ^* at an almost constant value of bypass ratio corresponds to region $n_{НД}$ = const.

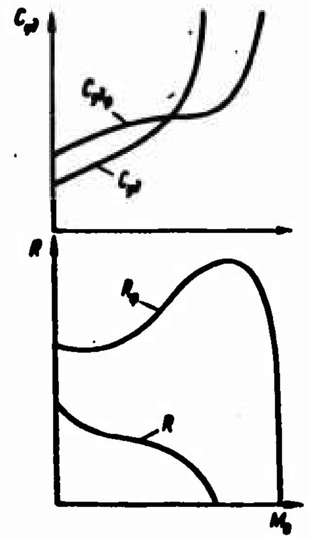


Fig. 4.38. Comparison of the high-speed characteristics of DTRD and DTRDF.

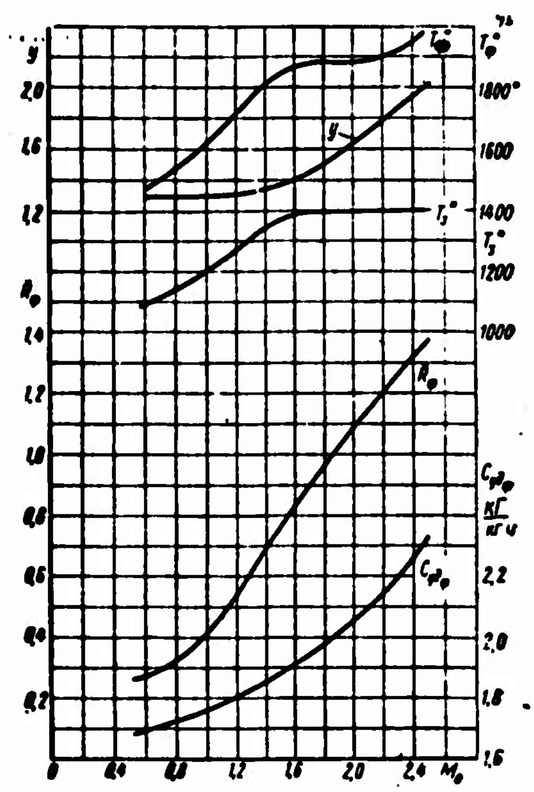


Fig. 4.39. High-speed characteristics of DTRDF^{II} at a combined program of control.

Constancy of parameters T_3^* and T_4^* and the growth of y correspond to region $n_{ВД}$ = const.

Thrust and specific fuel consumption of DTRDF[II] are continuously increased in the range of numbers M_0 = 0.6-2.5.

Peculiarities of High-Speed Characteristics of Two-Shaft Ducted-Fan Engines

Let us examine peculiarities of the high-speed characteristics of two-shaft ducted-fan engines with separate exhaust (diagram 1.3b) at two programs of control:

1) $n_{ВД}=\text{const}$ and $f_5=\text{const}$; 2) $n_{НД}=\text{const}$ and $f_5=\text{const}$.

Program of Control $n_{ВД}$ = const

With increase of the flight speed the bypass ratio of the engine rises since the flow rate of air through the secondary duct increases faster than through the primary duct ($\pi^*_{к_{II}} < \pi^*_{к_I}$). In this case the angles of incoming flow on the fan blades are increased, and are decreased on blades of the last stage of the high-pressure compressor.

If $\pi^*_{кВД} \simeq 6$, work of the high-pressure compressor maintains approximately a constant value with decrease of the given number of revolutions $n_{ВД(пр)}$. Then from the equation of work balance of the high-pressure turbocompressor

$$L_{кВД}=L_{тВД}\sim T_3^*$$

we find that the gas temperature before the turbine is also unchanged. Consequently, $T^*_{4(ВД)}$ = const.

Simultaneously there sets in "loading" of the fan and disturbance of the work balance of the low-pressure turbocompressor

$$(1+y)L_{\text{кнд}} > L_{\text{тнд}},$$

as a result of which the number of revolutions of the low-pressure turbocompressor shaft is lowered.

Program of Control $n_{\text{вд}}$ = const

In this case the loading of the fan at constant shaft revolutions leads to the fact that the automatic fuel metering control increases the fuel feed to the combustion chamber for a corresponding increase of work of the low-pressure turbine. Consequently, gas temperatures T_3^* and $T_{4(\text{вд})}^*$ grow.

The unbalance of works on the high-pressure turbocompressor ($L_{\text{твд}} > L_{\text{квд}}$), appearing because of the increase of T_3^*, is removed by spinup of the low-pressure turbocompressor rotor.

With climb to altitude the described regularities are changed to inverses. It is necessary to note that with observance of T_3^* = const the high-speed characteristics of single-shaft and two-shaft ducted-fan engines hardly differ from each other.

§ 4. Altitude Characteristics of Ducted-Fan Engines

Dependencies of total thrust, and also specific fuel consumption on the altitude at a constant flight speed (or M_0 = const) and the accepted program of engine control are called altitude characteristics of ducted-fan engines.

Let us consider the altitude characteristic of a single-shaft unboosted ducted-fan engine with program of control for maximum thrust n = const and T_3^* = const. The gas temperature before the turbine is kept constant by adjustment of the throat of the jet nozzle of the primary duct.

Construction and analysis of the altitude characteristic will be performed with usual assumptions:

1. $L_{\kappa}=\text{const}$. 2. $\eta^{*}_{\kappa}=\text{const}$, $\eta^{*}_{\tau}=\text{const}$, $\sigma^{*}_{\kappa.c}=\text{const}$,

$\varphi_{p.c}=\text{const}$, $\xi_{\kappa.c}=\text{const}$.

3. $p^{I}_{5}=p^{II}_{5}=p_{H}$.

As initial (stand) data let us take the same as during construction of the high-speed characteristic for a single-shaft ducted-fan engine. The M number at all flight altitudes is equal to 0.9.

Change of Specific Thrust (Fig. 4.40)

With climb to altitude at $n=\text{const}\,(L_{\kappa}=\text{const})$ the compression ratios of compressors in the ducts increase ($\pi^{*}_{\kappa_I}$ and $\pi^{*}_{\kappa_{II}}$) and the degree of preheating of working medium $\Delta=\frac{T^{*}_{3}}{T_{H}}$; is increased; this leads to an increase of the exit velocity from the jet nozzle of the primary duct. The exit velocity from the secondary duct, conversely, slowly decreases, since the temperature T^{*II}_{2} at the compressor outlet is lowered. As a result the specific thrust of the primary duct grows very considerably. The value of $R_{y\partial_{II}}$ is increased somewhat because of decrease of flight speed c_0 at M_0 = const. On the whole the specific thrust of a ducted-fan engine is increased considerably more intense than for an initial turbojet engine. Thus, at altitude 11 km we have $\bar{R}_{y\partial(ДТРД)}$ = 1.57, and $\bar{R}_{y\partial(ТРД)}$ = 1.34.

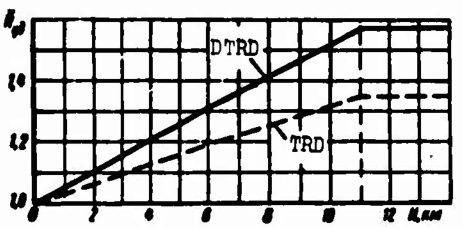


Fig. 4.40. Comparative change of the specific thrusts of ducted-fan and turbojet engines depending on the flight altitude.

Thus, the effect of altitude and speed of flight on the specific thrusts of ducted-fan and turbojet engines turns out to be opposite.

Change of Bypass Ratio of the Engine

Increase of the flight altitude leads to a drop of the airflow rate in the ducts. In this case the airflow rate is intensively decreased in the secondary duct, i.e., where the compression ratio of the compressor increases slower. As a result the bypass ratio of DTRD is lowered (Fig. 4.41). True, this lowering does not exceed 20% in the entire range of flight altitudes (up to H = 11 km).

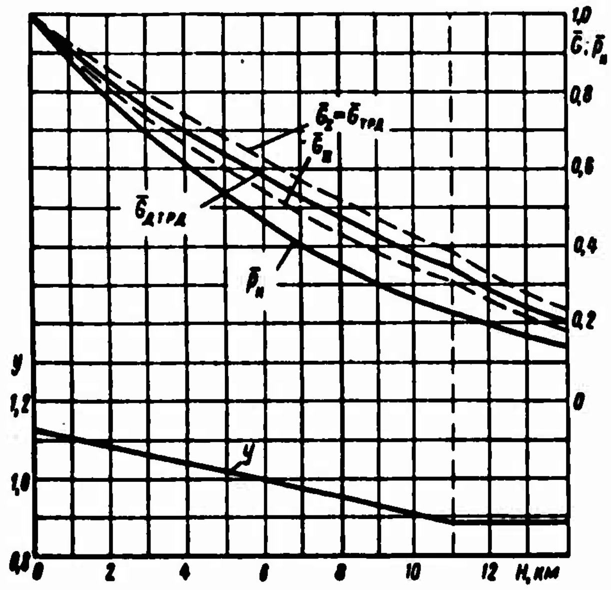


Fig. 4.41. Change of the airflow rate in ducts and the bypass ratio of a single-shaft ducted-fan engine with climb to altitude ($M_0=0,9$, $T_3^*=1200°$ K, $y=1$, $\pi^*_{к_{I(0)}}=15$, $\pi^*_{к_{II(0)}}=2,15$, $\pi_{р.с_I}=\pi_{р.с_{II}}$).

Decrease of the bypass ratio can be explained also by the fact that the airflow rate of the ducted-fan engine drops more intensively with climb to altitude than for the initial turbojet engine ($\bar{G}_{ТРД}=\bar{G}_I$). This may be seen from Fig. 4.41.

Change of Thrust of Ducted-Fan Engine

Specific thrust of a ducted-fan engine grows faster with increase of flight altitude than for the initial turbojet engine. The airflow rate of the ducted-fan engine is decreased, also more intense than for the turbojet engine. As a result the total thrust of a ducted fan engine is lowered somewhat slower than for a

turbojet engine in proportion to climb to altitude (Fig. 4.42).

Change of Engine Efficiency

Increase of the exit velocity from the jet nozzle with lowering of flight speed always leads to a drop of the thrust efficiency. Increase of the compression ratio and degree of preheating of the thermodynamic cycle somewhat improves the effective efficiency. As a result the overall efficiency of ducted-fan engine grows somewhat with climb to altitude, whereas at prescribed parameters of the working process for a turbojet engine it even slowly drops.

Change of the Specific Fuel Consumption (Fig. 4.42)

With increase of flight altitude the specific fuel consumption of a ducted-fan engine decreases. This is explained by more effective heat conversion into thrust work (i.e., growth of η_0) at some decrease of the work of each kilogram of thrust (because of lowering of c_0).

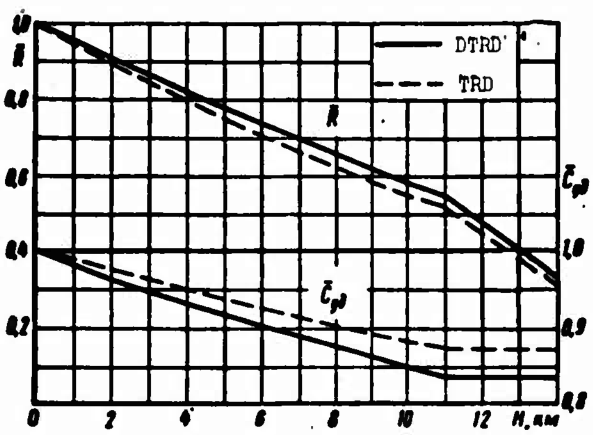


Fig. 4.42. Change of thrust and specific fuel consumption of a ducted-fan engine with climb to altitude ($M_0 = 0,9$, $T_3^* = 1200°$ K, $y = 1$, $\pi^*_{k_{I(0)}} = 15$, $\pi^*_{k_{II(0)}} = 2,15$, $\pi_{p.c_I} = \pi_{p.c_{II}}$).

Increase of the operating economy of a ducted-fan engine with climb to altitude is somewhat greater than for a turbojet engine (up to 4-5% at altitude 11 km).

§ . Peculiarities of Operational Characteristics of DTRD at Large Bypass Ratios

During the last two years the attention of scientists, researchers and designers working in the field of aviation has been attracted to the problem of designing ducted-fan engines of high thrusts with considerable bypass ratios (y = 6-8). At high values gas temperature before the turbine (T_3^* = 1300-1600°K) and total compression ratio π_k^* = 25-30 such ducted-fan engines can provide extremely low specific fuel consumption on the stand (C_{ya} = 0.28-0.35 kgf/kgf·h) and in flight at subsonic speed (C_{ya} = 0.56-0.65 kgf/kgf·h at number M_0 = 0.7-0.9 and H = 11 km). At present new prospective ducted-fan engines JT9-D1, RB-178, TF-39 are undergoing stand tests. At the working process parameters noted above these engines even turn out to be more economical than contemporary turboprop engines, besides considerably exceeding them by their high operational reliability, simplicity of construction, and low specific gravity ($\gamma_{дв}$ = 0.16-0.17 kgf/kgf).

The use of ducted-fan engines with large bypass ratios in civil aviation gives the possibility of considerably lowering the operational consumptions (by approximately 20-25%) and assuring a rapid growth of passenger shipments.

Below there are examined certain peculiarities of the operational characteristics of such engines.

Effect of the Bypass Ratio on the Drop of DTRD Thrust During the Takeoff Run

During the takeoff run on an airfield before takeoff the thrust of a turbojet engine (ducted-fan engine) always drops. This regularity is caused by a rapid increase of input pulse $\left(\frac{G}{g} V\right)$ with an output pulse that is practically constant for these aircraft speeds. For a turbojet engine the drop of thrust is small and does not exceed 5-7%. The drop of thrust of a ducted-fan engine is more considerable; it is greater, the smaller the absolute value

ratio of DTRD.

Relative thrust of TDRD during the takeoff run can be calculated by approximate formula

$$\bar{R}=1-\frac{V}{c_{5_{cp}}},$$

where $c_{5_{cp}}$ – the average gas exit velocity from the ducts.

Below are listed values $\bar{R}$ for liftoff speed of the aircraft $V_{отр}$ = 70 m/s at different bypass ratios:

y	0	1	2	3	4	6	8
$\bar{R}$	0,905	0,860	0,83	0,80	0,78	0,735	0,70

Drop of DTRD thrust on takeoff can be considered during the calculation of takeoff and landing characteristics of the aircraft.

Figure 4.43 shows the effect of y on the drop of DTRD thrust during the takeoff run.

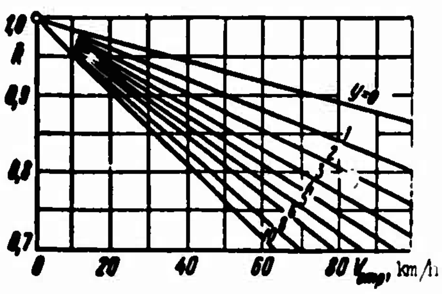


Fig. 4.43. Effect of the bypass ratio on the drop of DTRD thrust during the takeoff run.

Effect of the Bypass Ratio on Throttle Characteristics of the Engine at Cruise Flight Conditions

It is known that the value of engine thrust of a subsonic aircraft will be selected from the condition of providing satisfactory takeoff and landing characteristics. In flight at cruise performance (for example, at number $M_{0_{кр}}$ = 0.8 and $H_{кр}$ = 11 km) the thrust of

the propulsion system turns out to be excessive and it is necessary to throttle the turbojet engine (ducted-fan engine).

In Chapters II and IV it was shown that with lowering of the number of revolutions of TRD (DTRD) the specific fuel consumption is at first lowered and only with rough throttling does it start to increase. However, at large values of y throttling of a ducted-fan engine at cruise flight performance leads to the fact that lowering of $C_{y\pi}$ is delayed and is even ceased completely. When y > 6-8 the lowering of the number of revolutions of a two-shaft ducted-fan engine is even connected with a continuous rise of the specific fuel consumption (Fig. 4.44).

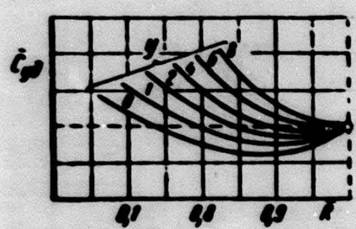


Fig. 4.44. Effect of the bypass ratio on the regularity of change of during DTRD throttling in flight.

Effect of the Bypass Ratio on the Takeoff Thrust of a Ducted-Fan Engine

Let us assume that the required aircraft thrust for flight at subsonic speed at altitude keeps a constant value. This means that the engine at cruise performance should also develop constant thrust. Then, with increase of the bypass ratio the takeoff (maximum) will rise, and the relationship of cruising (flight) thrust to takeoff (stand)

$$\frac{R_{\text{кр}(V,H)}}{R_{\text{взл}(0)}}$$

will continuously drop (Fig. 4.45).

When y = 8 as compared to y = 0 the takeoff thrust will increase approximately 60%, and relationship $\frac{R_{\text{кр}(V,H)}}{R_{\text{взл}(0)}}$ is reduced from 0.27 to 0.17.

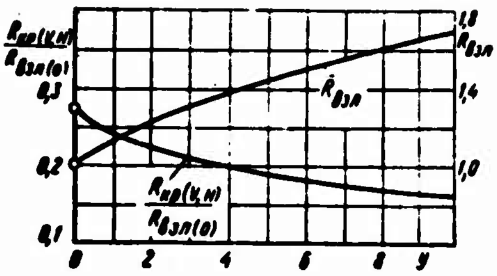


Fig. 4.45. Effect of the bypass ratio on the relationship of cruising flight and takeoff thrusts of a ducted-fan engine.

Sharper drop of DTRD thrust with climb to altitude in proportion to the growth of y is explained by the fact that at subsonic flight speeds (when $T_H^* < 288^\circ K$) the airflow rate drops faster and the specific thrust of the ducted-fan engine is increased slower. This is caused by the fact that $\pi^*_{\text{кII}}$ and, drops consequently, thrust of the secondary duct drops faster with the growth of y.

Increase of the takeoff thrust with growth of y improves the takeoff and landing characteristics of aircraft, but simultaneously leads to some loading of the propulsion system.

§ 6. Peculiarities of the Operational Characteristics of Ducted-Fan Engines with Mixing Chambers

At present the constructions of ducted-fan engines with mixing chambers (DTRD "Spey", TF-106, TF-30 and many others) are being increasingly applied. In these chambers there occur mass and power exchanges between flows exiting the compressor (fan) of the secondary duct and the turbine, installed in the primary duct (Fig. 1.3d).

After mixing or unification of flows in the common chamber the obtained mixture of gases, characterized by a specific degree of irregularity of velocity fields, total pressures and stagnation temperatures, proceeds either into the common afterburner or the common jet nozzle.

Numerous theoretical and experimental investigations showed that a rationally designed mixing chamber somewhat improves the engine economy. In an optimum case of mixing (when the unboosted ducted-fan

engine essentially differs in stagnation temperatures and has equal total pressures of initial flows) the specific fuel consumption is lowered by 1-2%.

Application of a mixing chamber gives the possibility of:

– simplifying the construction, control system and automatic outlet device of the engine, and lowering its weight;

– assuring the highest possible degree of thrust augmentation (application of separate boosting in the two DTRDF ducts is irrational, because of the difficulties of providing reliable cooling of the middle casing, both sides of which the hot gases flow around).

§ 7. Takeoff of Boosting DTRD

One essential advantage of the ducted-fan engines as compared to other types of gas-turbine aircraft engines is the possibility of very considerable thrust augmentation of the engine on takeoff by the additional combustion of fuel in the afterburners.

The biggest increase in DTRD thrust can be obtained practically with equality of boost temperatures and equal total pressures of gas in both ducts.

At constant total airflow rate ($G_{\Sigma} = \text{idem}$) the degree of boosting of DTRD depends only on the ratio of maximum boost temperature to the average mass temperature of gas in the ducts (before fuel combustion):

$$\frac{R^{I+II}_{ДТРДФ}}{R_{ДТРД}} = \sqrt{\frac{T^*_{Ф}}{T^*_{ср}}}, \qquad (4.24)$$

where

$$T^*_{ср} = \frac{T^*_{4} + yT_{п}}{1+y}.$$

With increase of y the degree of boosting continuously increases; when y = 2, T^{*}_{Φ} = 2000°K and T^{*}_{4} = 1000°K it is approximately equal to 2 (instead of 1.4 for a turbojet engine). Along with this, an increase of the bypass ratio decreases the optimum exit velocity of gas from the jet nozzles of the ducts and, consequently, the specific thrust of DTRD (DTRDF).

Thus, with growth of y when G_{Σ} = idem the ratio of total (specific) thrust of DTRDF (DTRD) to total (specific) thrust of TRDF (TRD) continuously drops (see Table 2).

Thus, the higher the degree of boosting of DTRD, the lower is its relative thrust as compared to TRDF. However, decreases of the latter is not so great as to seriously impede the aircraft takeoff conditions.

Table 2. Comparison of the degrees of boosting and relative thrusts of ducted-fan and turbojet engines (T^{*}_{Φ} = 2000°K, T^{*}_{4} = 1000°K, G_{Σ} = idem, H = 0, M_0 = 0).

Bypass ratio y	$\dfrac{R^{I+II}_{ДТРДФ}}{R_{ДТРД}}$	$\dfrac{R^{I+II}_{ДТРДФ}}{R_{ТРДФ}}$	$\dfrac{R_{ДТРД}}{R_{ТРД}}$
0	1,41	1,0	1,0
1	1,76	0,85	0,68
2	1,95	0,75	0,51
3	2,07	0,68	0,46

Let us note that a ducted-fan engine with a high bypass ratio with the afterburner turned off can provide especially good economy at subsonic speeds of flight, and with the afterburner turned on — a very great augmented thrust ratio.

Advantage of the ducted-fan engine with respect to the degree of boosting increases still more in flight.

§ 8. Operational and Technological Peculiarities of Ducted-Fan Engines

An important advantage of the ducted-fan engine (as compared to turbojet engine) is great fire safety. The external (secondary) engine duct is streamlined by comparatively cold air and has relatively low wall temperature. This prevents the danger of the appearance of fire from accidental drops of oil or kerosene getting on the heated external engine elements. The DTRD also possesses the advantage that its secondary duct represents unique "armor" or a shield from mechanical damage or striking of the most important elements of the construction, concentrated inside the basic duct (for example, combustion chambers, turbines, etc.).

Finally, the DTRD possesses great possibilities in the sense of an air bleed to blow off the boundary layer, the creation of "jet vanes." For this the tap of air from the secondary duct to the corresponding devices must be "organized." Losses of thrust with such bleed will be minimum.

Important questions, determining to a considerable degree the possibilities of introducing engines into aviation, make up the complex of industrial and technical problems. These include questions of design simplicity and manufacturability of the engine, its cost, duration of finishing of the experimental model to the series sample, etc.

Ducted-fan engines are considerably more complictaed in a constructive relationship than a single-spool turbojet engine. The problem of automation of regulation and control of this engine, especially in the region of high M_0 numbers of flight, is also resolved considerably more complicated than for a turbojet engine.

Finishing of the experimental sample of the ducted-fan engine to series production takes up considerably more time than for a turbojet engine. Therefore, it is extremely important even in the stage of preliminary designing to anticipate the possible modifications

and modernization of the engine, with its subsequent thrust augmentation, improvement of economy, etc.

There is still little accumulated design and industrial experience on the creation of ducted-fan engines. However, there are already convincing data showing that the tedious combined work of designers, technologists and metallurgists can lead to considerable simplification of the construction, reduction of prices of DTRD production, especially with a high bypass ratio increase of reliability in operation. In this respect the experiment of the English Rolls-Royce firm, having designed the RC_0-12 ducted-fan engine is very interesting. The rate of increase of service life of this engine is greater than for any other. On 1 April 1965 the engine service life was 7200 hours with intermediate inspection of the flame tubes. The number of ahead-of-schedule engine changes is minimum and composes 1 per 25,000 hours of nonfailure operating time.

Successful development of ducted-fan engines in the presence of favorable objective conditions will depend to a considerable degree on successful concrete solutions by the design collectives of plants and scientific research institutes of the entire complex of questions determining the effective aircraft power plant, and also on the critical evaluation of the accumulated experience of designing, finishing, production and operation of ducted-fan engines.

On Figs. 4.46 and 4.47 there are given basic operating characteristics of the ducted-fan engine "Conway" RC_0-12.

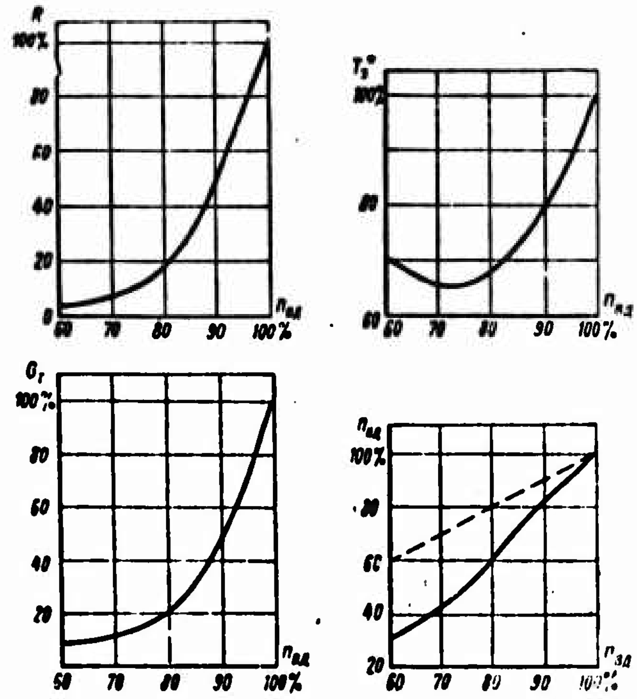


Fig. 4.46. Throttle characteristics of the ducted-fan engine "Conway" RC_0-12.

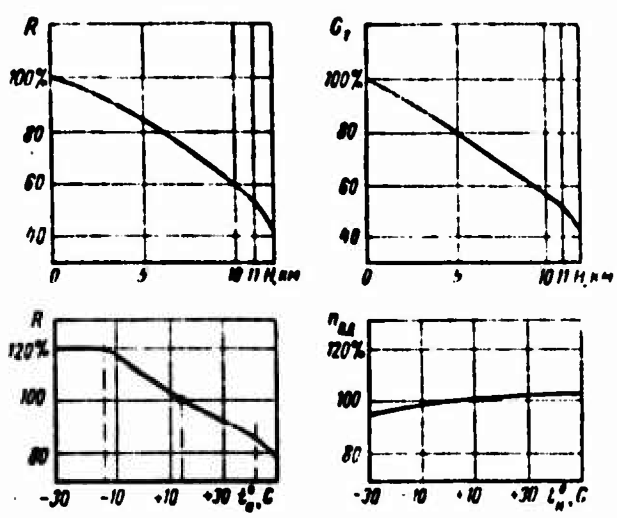


Fig. 4.47. Effect of the external temperature and flight altitude on parameters of the "Conway" RC_0-12 ducted-fan engine.

CHAPTER V

OPERATIONAL CHARACTERISTICS OF TURBOPROP ENGINES

§ 1. Throttle Characteristics of Turboprop Engines

Dependences of effective power, effective fuel consumption, and also the reaction thrust on the degree of fuel feed throttling to the engine at a prescribed program of control are called turboprop engine throttle characteristics. Since change of fuel feed is usually connected with change of the number of revolutions of the engine shaft, the throttle characteristics are depicted as curves of the dependence of basic turboprop engine parameters N_e, C_e and R on the number of revolutions of the engine shaft.

In bypass turboprop engines the throttle characteristics are depicted as functions of turbocompressor shaft revolutions (or high-pressure stage). In cases when the throttle characteristics are taken at a constant number of revolutions, they are depicted in the form of functions of fuel consumption per hour:

$$N_e = f_1(\bar{G}_T), \quad C_e = f_2(\bar{G}_T), \quad R = f_3(\bar{G}_T).$$

On the throttle characteristics of turboprop engines there is also plotted a curve of the change of gas temperature before the turbine or behind it. This curve gives the possibility of judging the thermal state of the "hot" part of the engine during its operation.

Throttle characteristics are usually obtained experimentally, on a stand. They can also be obtained analytically with a high degree of accuracy.

Throttle Characteristics of Single-Shaft Turboprop Engines

Let us examine the throttle characteristics of a single-shaft turboprop engine (Fig. 5.1). Let us assume that at nominal (takeoff) conditions in the turbine of the turboprop engine there occurs total expansion of gas: $p_4' = p_5 = p_H$.

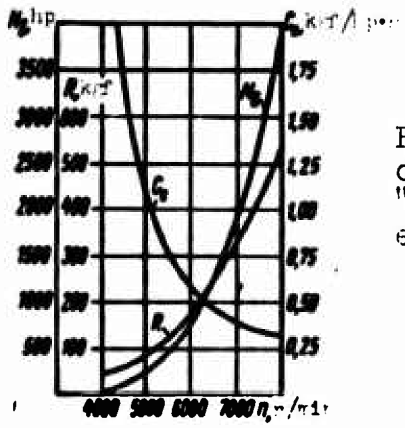


Fig. 5.1. Throttle characteristic of "Mamba" turboprop engine.

Such a case of operation of the turbine of a turboprop engine is typical.

With decrease of fuel feed to the combustion chamber the gas temperature before the turbine is lowered. In this case the turbine power drops, due to which the power balance is disturbed: $N_т < N_к + N_в$. As a result the engine shaft revolutions are decreased.

With decrease of the engine shaft revolutions constant propeller blade angle the propeller output and equivalent power of the turboprop engine drop, the reaction thrust of the turboprop engine is also decreased, and the effective fuel consumption continuously rises. The gas temperature before the turbine (and past it), having a high initial value under takeoff conditions, at first drops

with lowering of revolutions, reaching some minimum value at medium revolutions, and then is continuously increased.

For explanation of the regularity of the course of basic turboprop engine parameters with respect to the number of revolutions let us preliminarily examine how the temperature and pressure of gas are changed in characteristic sections of the engine during throttling.

<u>Change of the Gas Temperature and Pressure in Characteristic Sections of the Gas-Air Duct of the Turboprop Engine.</u>

Figure 5.2 shows the change of compression ratio of the compressor and expansion ratio of the turbine depending on the number of revolutions. Since the exit velocity of gas from the exhaust of the turboprop engine is small (it is somewhat less than the absolute velocity at the turbine exit), then the value of $\pi^*_т$ hardly differs from $\pi^*_к$. With throttling of the turboprop engine it drops continuously. Continuous lowering of the value of $\pi^*_т$ leads to the fact that the gas temperature before the turbine, depending on the revolutions for a turboprop engine, in the entire range

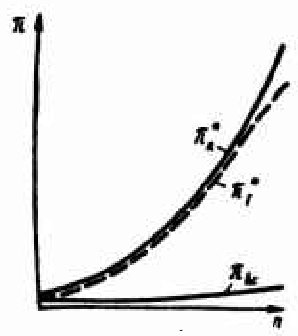


Fig. 5.2.

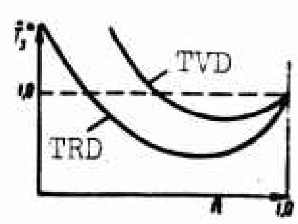


Fig. 5.3.

Fig. 5.2. Change of $\pi^*_к$, $\pi^*_т$ and $\pi_{в.с}$ of a single-shaft turboprop engine depending on the number of revolutions.

Fig. 5.3. Change of $\bar{T}^*_3$ of single-shaft turboprop and turbojet engines depending on the number of revolutions.

of working revolutions turns out to be considerably higher than for the turbojet engine at equal values of $\pi^*_{\kappa(p)}$ and $T^*_{3(p)}$ at initial nominal (or takeoff) conditions (Fig. 5.3).

Actually, by solving the equation of work balance of the turbocompressor

$$L_т = 118 T^*_3 \left(1 - \frac{1}{\pi_т^{*0,25}}\right) \eta^*_т = L_к + L_в \qquad (5.1)$$

with respect to T^*_3, we obtain

$$T^*_3 = \frac{Cn^2}{118\left(1 - \frac{1}{\pi_т^{*0,25}}\right) \eta^*_т}, \qquad (5.2)$$

where $C \simeq \text{const.}$

From equation (5.2) it follows that continuous lowering of $\pi^*_т$ shifts dependence $T^*_3 = f(n)$ for the turboprop engine to the region of raised values of gas temperatures as compared to case $\pi^*_т = \text{const}$ for turbojet engine.

Here we approximately considered that the specific work of the propeller, as the work of the compressor, is changed in proportion to the square of revolutions.

Change of Power Depending on the Number of Revolutions.

Numerous tests of turboprop engines show that the law of change of propeller output of the turboprop engine with respect to the number of revolutions depends on the propeller blade angle (φ^o).

When φ^o = const the dependence of propeller output on the number of revolutions is depicted rather exactly by cubic parabola

$$N_в = An^3, \qquad (5.3)$$

where A = f(φ^o).

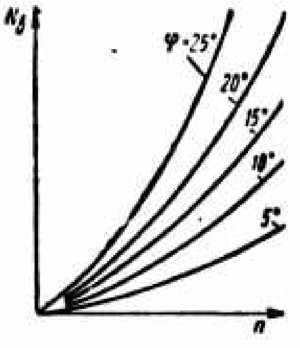


Fig. 5.4. Effect of the propeller blade angle φ on curves N_B.

At a constant number of revolutions with increase of the blade angle (with loading of propeller) the propeller output grows, with decrease of the angle (lightening of propeller) it is reduced. Thus, different cubic parabolas N_B, passing through the origin of coordinates, correspond to different values of angle φ (Fig. 5.4).

Let us clarify how the available power of the turbine of a turboprop engine is changed with change of angle φ.

If at constant engine revolutions the pilot loads the propeller, having increased angle φ, there will be disturbed the balance of powers

$$N_T < N_K + N_B$$

and the turboprop engine revolutions will start to be reduced. But the speed regulator, trying to keep the revolutions constant, will restore the disturbed equilibrium by increase of the fuel feed to the combustion chamber (with lowering of the angular velocity of rotation the centrifugal weights of the regulator will transfer throttling needle to increase of G_T). As a result the gas temperature before the turbine will increase, and the turbine power will be increased in accordance with increase of propeller output.

Thus, the family of curves T_3^* and C_e will correspond to the family of curves N_B for various values of φ^0. The greater φ is, the higher the gas temperature before the turbine T_3^* and the lower the effective fuel consumption C_e. Actually, when n = const loading of the propeller leads to increase of T_3^* and π_K^*, which

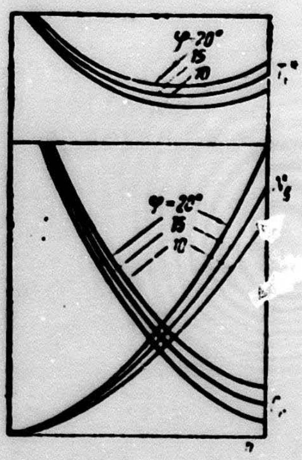


Fig. 5.5. Throttle characteristics of a turboprop engine at different propeller blade angles.

increases the effective efficiency of the engine and lowers the effective fuel consumption. This follows directly from formula

$$C_e = \frac{632}{H_u \eta_{ie}}. \qquad (5.4)$$

Thus, to each value of φ = const there corresponds a specific regularity of flow of working process parameters and output data of the turboprop engine with respect to the number of revolutions (Fig. 5.5).

Since the airflow rate is changed rather exactly by the linear dependence on the number of revolutions, i.e.,

$$G_a \sim n,$$

in accordance with equation (5.3) it may be concluded that when φ = const

$$L_a \sim L_e \sim n^2, \qquad (5.5)$$

i.e., propeller work turns out to be proportional to the revolutions squared.

Causes leading to the growth of $L_a \cong L_e$ with increase of turboprop engine revolutions are basically the same as for a turbojet engine, i.e., increase of the compression ratio of compressor at relatively small variable value of T_3^*.

Change of C_e Depending on the Number of Revolutions.

Continuous lowering of the effective fuel consumption with increase of revolutions of a single-shaft turboprop engine is explained mainly by the growth of π_{k}^{*}. This promotes a considerable increase of gas temperature T_3^* in the region of high revolutions.

Thus, conditions of maximum propeller output of a turboprop engine coincides with conditions of its best economy. Any throttling of the turboprop engine is connected with impairment of the operating economy of the engine. In this respect the properties of throttle characteristic of a turboprop engine differ from a turbojet, during throttling of which the specific fuel consumption is at first lowered.

Effect of Adjustment of the Propeller Blade Angle of the Throttle Characteristics of a Single-Shaft Turboprop Engine.

Let us examine how propeller adjustment affects the throttle characteristics of a single-shaft turboprop engine (Fig. 5.6).

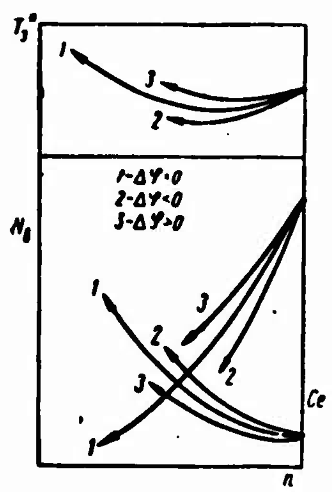


Fig. 5.6. Effect of adjustment of angle φ on the throttle characteristic of a single-shaft turboprop engine.

Let us assume that curves 1 correspond to a certain constant propeller blade angle (φ = const). With lightening of the propeller during throttling, power N_B drops, and the effective fuel consumption

increases sharper (curves 2) than when φ = const. Conversely, with loading of the propeller, (curves 3) the corresponding throttle characteristics are flatter. Figure 5.6 also shows the effect of propeller adjustment on the regularity of change of T_3^* depending on the number of revolutions.

By analyzing the curves on Fig. 5.6, at first glance it seems expedient to always load the propeller when throttling the turboprop engine. However, this is not entirely correct. For example, during engine starting a very important property is its ability to be rapidly brought to maximum power. For all possible improvement of turboprop engine pickup it is necessary that the excess turbine power (at $T^*_{3(\text{макс})}$) over the total power of the compressor and propeller be the greatest. For this purpose during starting it is necessary to lighten the propeller as much as possible, setting it at an angle close to zero power angle. Then

$$\Delta N_{\text{т}} = N_{\text{т(макс)}} - N_{\text{к}} = \text{maximum}.$$

For a number of contemporary single-shaft turboprop engines the engine starting and advance to maximum revolutions are accomplished when φ = 0° (Fig. 5.7a). This means that control of the operating conditions of a turboprop engine in operation occurs when n = const. Such a control system provides good engine pickup.

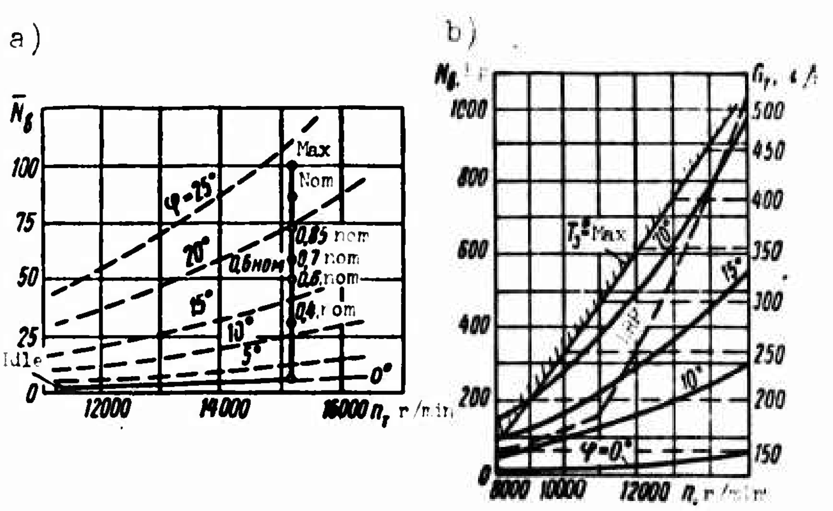


Fig. 5.7. Real law of change of N_e and φ depending on the number of revolutions of turboprop engines: a) single-shaft turboprop engine (N_e = = 4000 hp); b) turboprop engine "Mamba."

On Fig. 5.7b there is shown the law of change of blade angle depending on the revolutions for turboprop engine "Mamba." From the figure it is clear that acceleration of the engine in the range of revolutions $\Delta n = (0{,}2 - 0{,}8)\, n_{макс}$ occurs with minimum angle $\varphi = 12^{\circ}$. Only in the range $\Delta n = (0{,}8 - 1)\, n_{макс}$ does there occur loading of the propeller with increase of φ from 12 to 23°.

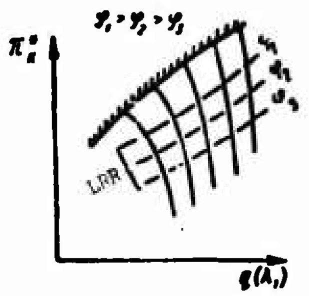


Fig. 5.8. Effect of adjustment of φ on the line of operating conditions of a single-shaft turboprop engine.

Figure 5.8 shows a family of lines of operating conditions of the turbocompressor of a single-shaft turboprop engine when φ = const. With loading of the propeller, line φ = const is displaced to the region of raised values of T_3^* and $\pi_к^*$; in this case the surge margin of the compressor is reduced.

Reduction of Turboprop Engine Parameters to Standard Atmospheric Conditions

Reduction of turboprop engine parameters to standard atmospheric conditions is carried out at a fixed propeller pitch (φ = const).

Reduction formulas for thrust, number of revolutions, airflow rate and fuel consumption for turbojet and turboprop engines are identical.

<u>Reduction Formula for Effective (Propeller) Power.</u>

We have

$$N_э \sim L_в G_в,$$

where

$$L_{в} \sim T^*_H, \quad G_{в} \sim \frac{p^*_H}{\sqrt{T^*_H}}.$$

Then

$$N_{в} \sim p^*_H \sqrt{T^*_H}.$$

Hence it is easy to obtain the reduction formula for a turboprop engine

$$N_{в_{пр}} = N_{в_{изм}} \frac{760\sqrt{288}}{p_0\sqrt{T_0}}. \quad (5.6)$$

Effective fuel consumption

$$C_e = \frac{632}{H_u \eta_e} \left[\frac{\text{kgf}}{\text{h.ph}}\right]$$

does not need reduction to standard atmospheric conditions, since under similar conditions it keeps a constant value (η_e = const).

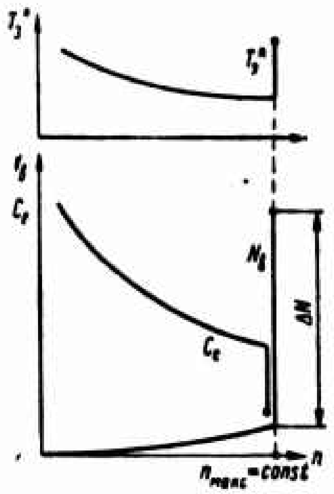
Throttle Characteristic of a Turboprop Engine When n = const

On Fig. 5.9 there is depicted the basic throttle characteristic of single-shaft turboprop engine when $n = n_{макс} = \text{const}$. Similar characteristic of a typical single-shaft turboprop engine, constructed in relative parameters, is presented on Fig. 5.10.

Throttle Characteristics of Two-Shaft Turboprop Engines

Let us examine the throttle characteristics of two-shaft turboprop engines. As before we will assume that under all engine operating conditions there is observed total expansion of gas in the turbine $p_4' = p_5 = p_H$.

Figure 5.11 shows the throttle characteristic of two-shaft turboprop engine "Tyne" RTy-12 with a split compressor. As can be



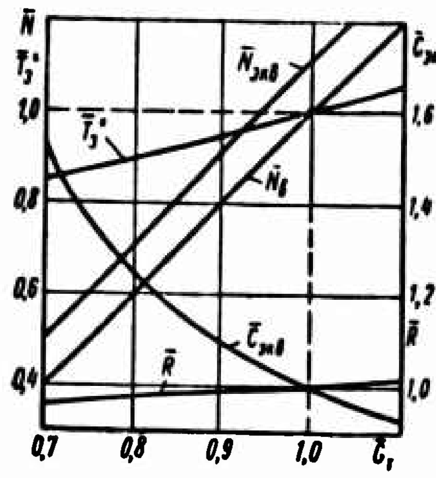
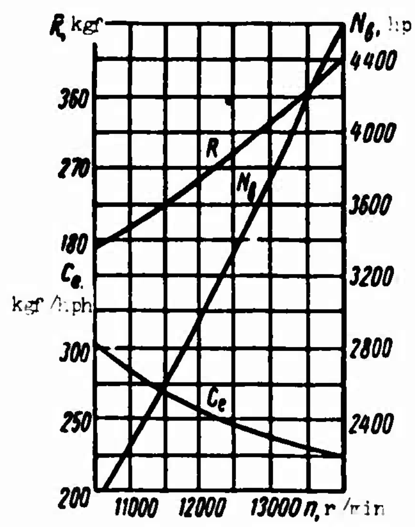


Fig. 5.9.   Fig. 5.10.

Fig. 5.9. Throttle characteristic of a turboprop engine when n = const.

Fig. 5.10. Throttle characteristic of single-shaft turboprop engine in relative parameters.



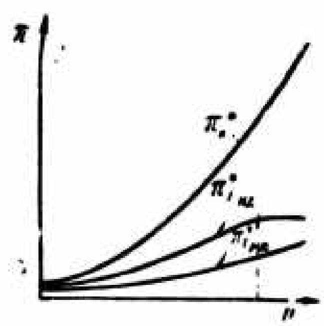


Fig. 5.11.   Fig. 5.12.

Fig. 5.11. Throttle characteristic of turboprop engine "Tyne" RTy-12 with split compressor.

Fig. 5.12. Change of $\pi^{*}_{\text{т ВД}}$ and $\pi^{*}_{\text{т НД}}$ of a two-shaft turboprop engine depending on the number of revolutions.

seen, with decrease of the number of revolutions of the high-pressure turbocompressor the propeller output and the reaction thrust are lowered as before, and effective fuel consumption is increased. Thus, qualitative changes of the basic turboprop engine parameters (N_e, R and C_e) coincide during throttling of single-shaft and two-shaft engines.

In order to clarify quantitative distinctions of these characteristics, let us examine the regularity of change of gas pressure and temperature drops in the basic elements of two-shaft turboprop engines depending on the number of revolutions.

Change of the Gas Temperature and Pressure in Characteristic Sections of a Two-Shaft Turboprop Engine (Fig. 1.4b).

Figure 5.12 contains curves of change of the compression ratio of compressor and the expansion ratios of gas in high-pressure and low-pressure turbines. From the figure it is clear that during throttling in a certain range of revolutions the high-pressure turbine turns out to be "blocked" with respect to pressure drop (as long as in the primary nozzle box of the low-pressure turbine there are preserved critical outflow conditions). The law of change of gas temperature T_3^* depending on the number of revolutions is determined by equation of work balance of high-pressure turbocompressor

$$L_{т_{вд}} = L_{к_{вд}},$$

whence we find

$$T_3^* = \frac{L_{к_{вд}}}{118\left[1-\frac{1}{\pi_{т_{вд}}^{*0.25}}\right]\eta^*_{т_{вд}}}.$$

Now it is easy to arrive at the conclusion that with equal values of $\pi^*_{к_{вд}}$ and T_3^* under calculated conditions the regularity of change of gas temperature before the turbine T_3^* for two-shaft turbojet, turboprop and ducted-fan engines is practically the same.

Effect of Regulating the Propeller Pitch on the Throttle Characteristic of a Two-Shaft Turboprop Engine.

It is possible to consider with a high degree of accuracy that in a turboprop engine with total gas expansion in the low-pressure turbine

$$L_{в}=L_{т_{НД}}\simeq L_{e}.$$

Let us examine the peculiarities of the throttling process of a two-shaft turbojet engine.

Let us assume that φ = const. With decrease of fuel feed to the combustion chamber because of lowering of gas temperature T_3^* and the appearing work unbalance of the high-pressure turbocompressor the revolutions of the latter drop. However, the power of the low-pressure turbine is reduced faster than the high-pressure turbine, since, besides decrease of T_4^* and $G_в$ pressure differential $\pi^*_{т_{НД}}$ still drops. Therefore, the appearing work unbalance of the low-pressure turbocompressor is eliminated by more intense lowering of the angular spin velocity of the low-pressure stage.

How does regulating the propeller pitch affect the throttle characteristic of a two-shaft turboprop engine?

First of all let us note that regulating the propeller pitch practically does not affect the power and work of the low-pressure turbine, the working process parameters of which are determined completely by the high-pressure turbocompressor. Therefore, regulating the propeller pitch leads only to a change of the number of revolutions of the low-pressure turbocompressor so that, as before, there is observed condition:

$$N_{в}=N_{т_{НД}} \text{ or } L_{в}=L_{т_{НД}}.$$

With change of angle φ there are changed the number of revolutions of the low-pressure compressor, propeller efficiency $\eta_{в}$, low-pressure turbine efficiency $(\eta^*_{т_{НД}})$ and, accordingly, propeller

output $N_в$ and thrust $P_в$, i.e.,

$$N_в=\frac{G_вL^*_{аз\,т\,НД}}{75}\eta^*_{т\,НД}\sim\eta^*_{т\,НД},\qquad (5.7)$$

whence

$$P_в=\frac{75N_в\eta_в}{c_0}\sim\eta^*_{т\,НД}\eta_в.\qquad (5.8)$$

Thus, control of angle φ gives the possibility, after having established the most advantageous low-pressure stage revolutions, of assuring the maximum values of ($\eta^*_{т\,НД}$) and $\eta_в$, and consequently, $N_в$ and $P_в$ (Fig. 5.13).

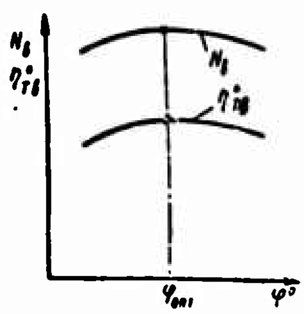


Fig. 5.13. Effect of regulating the propeller blade angle on the propeller output of a two-shaft turboprop engine.

Let us note also that regulating the propeller pitch in a two-shaft turboprop engine does not affect the position of the line of operating conditions of the compressor characteristic.

Thus, throttle characteristics of a two-shaft turboprop engine differ from similar characteristics of a single-shaft turboprop engine by the regularity of change of T^*_3 depending on the number of revolutions (which has a known effect on the shape of curves $N_п$, R and C_e), and also by the possibility of providing optimum values of efficiency of the propeller and low-pressure turbine (maximum of the product of these efficiencies) with the help of a system of two separate turbines.

If it is possible to provide optimum law of change of T^*_3 depending on the number of revolutions by control of φ in a single-shaft turboprop engine, then the possibility of achievement of the

highest values of $\eta_{в}$ and $\eta_{т}^{*}$ nevertheless remains an advantage of the two-shaft turboprop engine.

§ 2. <u>High-Speed Characteristics of Turboprop Engines</u>

The dependences of propeller output or total (equivalent) power, and also the effective fuel consumption on the flight speed at an assigned program of control are called the <u>high-speed characteristics</u> of a turboprop engine.

The high-speed characteristic of a turboprop engine can be obtained experimentally, for example in flight tests with the aid of a flying laboratory, and also approximately by the analytic method.

Programs of Control of Turboprop Engines in Flight

The given programs include programs of control for maximum propeller output and maximum total (equivalent) power, for best economy and others. Each of these programs assures a change of the working process parameters, at which the assigned dependence of change of $N_{в}$, $N_{экв}$ or C_e is realized automatically at all flight speeds and altitudes.

Controllable Parameters and Controlling Factors in Turboprop Engines

A turboprop engine generally has more performance parameters than the usual turbojet engine, and accordingly more controlling factors and elements. For example, for a single-shaft turboprop engine an additional controlling factor as compared to a turbojet engine[1] is the propeller blade angle φ°, and an additional control element — prop regulator.

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[1]A single-shaft turbojet engine with fixed geometry is considered.

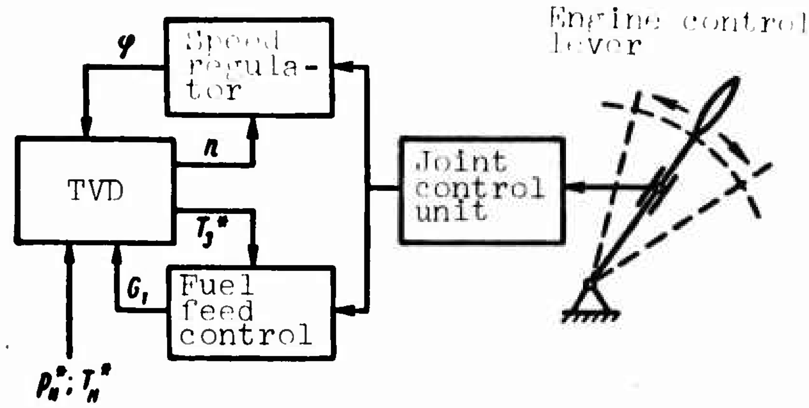

Fig. 5.14. Block diagram of single-shaft turboprop engine control.

Control of angle φ for a turboprop engine fulfills the same role as control of the throat (exit section) of the jet nozzle of a turbojet engine and permits changing the number of engine revolutions and the gas temperature before the turbine independently of each other.

On Fig. 5.14 there is shown a block diagram of control of a single-shaft turboprop engine with two independent regulators: rpm and fuel feed. The engine has a joint control unit for regulators[1] with a single engine control lever. The connection between controlling factors and controllable parameters is carried out by scheme

$$G_т \rightarrow T_3^*; \quad \varphi \rightarrow n.$$

In a two-shaft turboprop engine the additional controlling factor, as before, turns out to be angle φ, and the controllable parameter — number of revolutions of propeller shaft $n_в$ or low-pressure stage ($n_{нд}$). In this case the number of revolutions of the high-pressure turbocompressor $n_{вд}$ and the temperature is controlled, as for a simple turbojet engine, by change of the fuel feed by an automatic fuel feed control, interlinked with the regulator. The number of revolutions of the low-pressure turbocompressor is

[1]Command-fuel unit (KTA).

controlled by the second, independent of the first, speed regulator.

The connection between controlling factors and controllable parameters of a two-shaft turboprop engine is carried out by scheme

$$G_{\text{т}} \begin{cases} n_{\text{тк}} \\ T_3^* \end{cases} ; \quad \varphi \rightarrow n_{\text{в}}$$

Figure 5.15 contains a block diagram of two-shaft turboprop engine control.

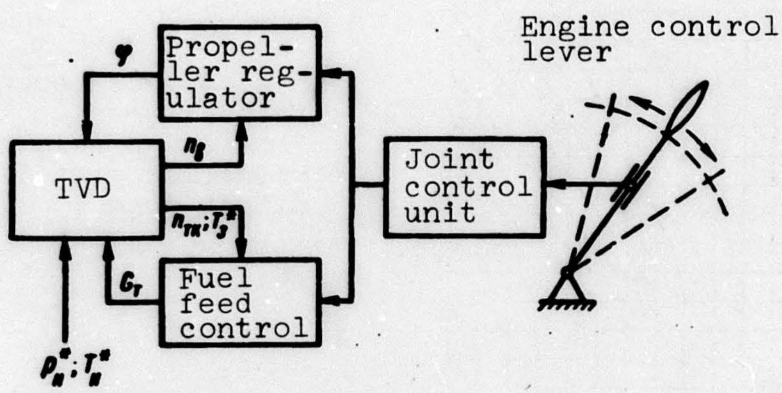

Fig. 5.15. Block diagram of two-shaft turboprop engine control.

The engine can be controlled by one lever, movement of which at all flight speeds and altitudes gives the possibility of automatically maintaining maximum product of $\eta_{\text{в}} \eta^*_{\text{тНД}} = \max$.

The introduction of a jet nozzle regulator (f_5 = var) for a turboprop engine gives the possibility of providing the most advantageous distribution of energy between the propeller and reaction and, consequently, maximum equivalent (total) horsepower. In accordance with the principle of redistribution of turboprop engine thrust as the flight speed increases the jet nozzle should simultaneously be covered for increase of jet thrust and the propeller should be lightened for lowering of the propeller thrust, while observing condition n = const and T_3^* = const. In this case the lightening of the propeller and, consequently, lowering of $L_{\text{в}}$ will

be compensated by a decrease of the pressure drop on the common turbine in order to continuously fulfill equality

$$L_{\text{т}}=L_{\text{к}}+L_{\text{в}}.$$

The expediency of introduction of an exhaust nozzle regulator is determined by the effect of change of the exit velocity c_5 on the thrust (or equivalent horsepower) of a turboprop engine.

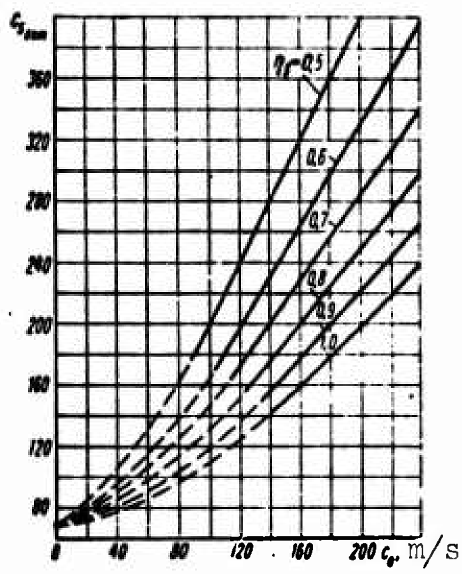

Fig. 5.16. Dependence of optimum exit velocity from the nozzle on the propeller efficiency and flight speed.

Figure 5.16 shows the dependence of optimum exit velocity from the nozzle on the propeller efficiency and flight speed. From the figure it is clear that with increase of flight speed from 0 to 220 m/s the optimum exit velocity at $\eta_{\text{в}}=0{,}6$ increases from 70 to 360 m/s.

It was noted above that on a stand the deviation of exit velocity $c_5 \approx c_{4a}$ from optimum value leads to a considerable drop of thrust. With increase of flight speed this effect weakens more and more (Fig. 5.17). At high transonic flight speeds the deviation of c_5 from optimum value practically scarcely affects the turboprop engine thrust. Thus, control of the turboprop engine exhaust nozzle would make sense mainly on a stand, and also at low and moderate flight speeds.

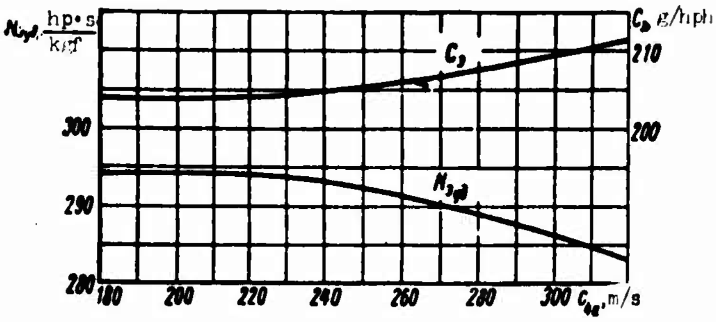

Fig. 5.17. The effect of gas exit velocity from the turbine on the specific equivalent horsepower and equivalent fuel consumption in flight.

However, it should be considered that the introduction of a variable-area nozzle will require structural complication, increase of dimensions and weight of the engine, which are not justified without proper compensation in thrust and economy. Therefore, in practice a turbine with total gas expansion and fixed-area exhaust device is applied. For such turboprop engines some losses of thrust occur on the stand, i.e., under conditions when the relative thrust of the turboprop engine (as compared to turbojet) reaches maximum value.

High-Speed Characteristic of Single-Shaft Turboprop Engine

Let us examine the high-speed characteristic of a single-shaft TVD, designed for constant flight altitude and program of control for maximum thrust.

Program of control of the turboprop engine for maximum total thrust or maximum total (equivalent) horsepower includes the following conditions:

$$1)\ n = n_{макс} = \text{const};$$

$$2)\ T_3^* = T^*_{3(макс)} = \text{const};$$

$$3)\ x = x_{опт} = f(c_0)\ \text{or}\ c_{5(опт)} = \frac{c_0}{\eta_в}.$$

Conditions 1) and 2) are fulfilled by using a prop regulator and fuel feed control. Observance of condition 3) requires the introduction of a jet nozzle regulator; therefore it can be replaced by condition

$$3') \; f_5 = \text{const and } p_4^* = p_5 = p_H.$$

At flight speeds above 600-700 km/h the realization of condition 3' instead of 3 gives little difference in results and practically assures maximum thrust of the turboprop engine.

Basic Assumptions.

We will assume that: 1) when n = const we have $L_K = \text{const}$, and 2) particular loss factors, efficiency of the turbine and compressor keep constant value, i.e., $\eta_T^* = \text{const}$, $\eta_K^* = \text{const}$, $\sigma_{BX}^* = \text{const}$, $\sigma_{K.c}^* = \text{const}$, $\varphi_{B.c} = \text{const}$, $\xi_{K.c} = \text{const}$.

Let us now examine the peculiarities of the high-speed characteristic of a turboprop engine.

With increase of flight speed the total compression ratio rises and accordingly the expansion ratio in the turbine; the exit velocities at the turbine exit and the turboprop engine exhaust pipe also rise. Really, from the flow rate equation written for the throat of the primary nozzle box of the turbine and the exhaust pipe exit section we obtain

$$\pi_T^{*\frac{n+1}{2n}} = \frac{f_5 q(\lambda_5)}{f_{CA} q(\lambda_{CA})} \sim q(\lambda_5). \qquad (5.9)$$

From this it follows that with a growth of $q(\lambda_5)$ there is increased

From the equation of work balance of turbocompressor

$$L_T = L_K' + L_B$$

the conclusion can be made that for preservation of $T_3^* = \text{const}$ with increase of the M_0 number of flight, and consequently, increase of

the total work of the turbine L_T, it is necessary to load the propeller. In this case the excess work of the turbine is completely consumed by the propeller.

Change of $N_в$ Depending on the Flight Speed.

With increase of flight speed the airflow rate $G_в$, continuously increases by the same regularity as for a turbojet engine. Specific work of the propeller $L_в$ also increases due to increase of the pressure drop on the turbine (Fig. 5.18). Therefore, with increase

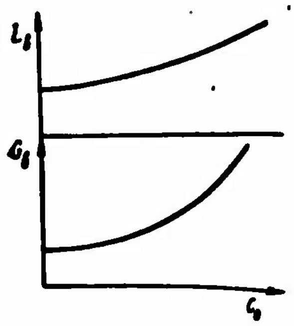

Fig. 5.18.

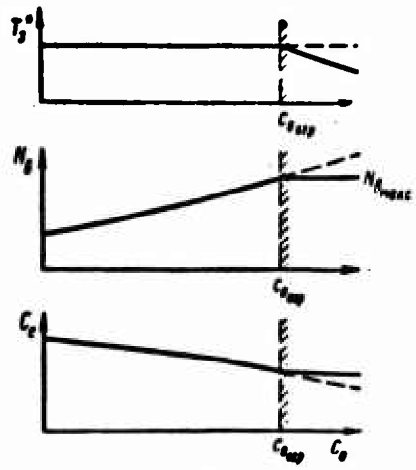

Fig. 5.19.

Fig. 5.18. Change of $G_в$ and $L_в$ depending on the flight speed.

Fig. 5.19. Change of propeller output and effective fuel consumption of a turboprop engine depending on the flight speed.

of flight speed the propeller output of a turboprop engine continuously increases (Fig. 5.19). Increase of $N_в$ at flight speed c_0 = 200-250 m/s is very considerable and can be 30-50% of the initial value of power when c_0 = 0.

Change of C_e Depending on the Flight Speed.

From expression $C_e = 3600\frac{m_T}{N_{в_{уд}}} \sim \frac{T_3^* - T_2^*}{L_в}$ it follows that as the flight speed increases the effective fuel consumption of the

turboprop engine is intensively decreased (see Fig. 5.19). Really, the relative fuel consumption per 1 kgf of air continuously drops, and specific work of the propeller increases. Lowering of C_e depending on the flight speed is explained thermodynamically by increase of η_e because of the increase of π.

Lowering of C_e at flight speed c_0 = 200-250 m/s can be 15-25% of the initial value of fuel consumption when c_0 = 0.

Change of Reaction Thrust.

With increase of flight speed the gas exit velocity at the turbine exhaust rises. However, specific reaction thrust

$$R_{y\text{д}} \sim (c_5 - c_0)$$

drops in this case, more intensively than for the turbojet engine (because of smaller initial values of $c_{5\ (0)}$). Calculations show that total reaction thrust also drops (Fig. 5.20). Lowering of

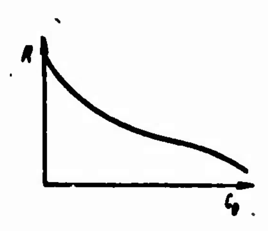

Fig. 5.20. Change of reaction thrust of a turboprop engine depending on the flight speed.

reaction thrust at flight speed c_0 = 200-250 m/s can be 20-30% of its takeoff value.

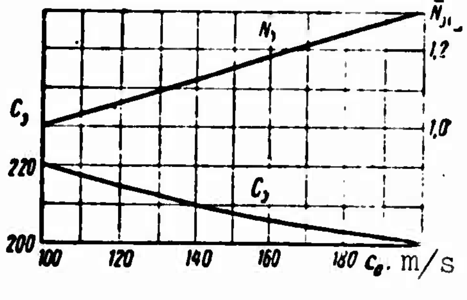

Fig. 5.21. High-speed characteristic of a single-shaft turboprop engine (altitude 8000 m).

Figure 5.21 shows the high-speed characteristic of a single-shaft turboprop engine.

<u>Limitation of Propeller Output Depending on the Flight Speed.</u>

Continuous increase of the propeller output of a turboprop engine leads to increase of loads (torque, circumferential forces) on components of the propeller gear. Safeguard of reliable operation of this extremely important structural unit requires either increase of the gear weight (for greater strength of components), or limitation of the amount of propeller output, starting from the flight speed at which $N_в$ reaches a prescribed limiting value. The propeller output is limited by torque on the turbine shaft by means of lowering the turboprop engine revolutions or by decreasing the gas temperature before the turbine (for example, lightening of the propeller). But lowering of T_3^* leads to deterioration of the operating economy of the engine under flight conditions with limitation (Fig. 5.19).

Peculiarities of the High-Speed Characteristics of Two-Shaft Turboprop Engines

Let us examine the peculiarities of the high-speed characteristics of two-shaft turboprop engines (see Fig. 1.4b).

Let us assume that the program of control of the engine includes the following conditions:

1) $n_{тк}=n_{макс}=\text{const}$;

2) f_5 = const;

3) $n_в=n_{нанв}$ (from condition of assuring $\eta^*_{тв}\,\eta_в=\text{макс}$).

In a case when at $n_{тк}=\text{const}$ we have $L_к=\text{const}$ and $\pi^*_{т_{вд}}=\text{const}$, conditions f_5 = const and $T_3^*=\text{const}$ are equivalent.

Let us assume, as before, that at all flight speeds $p_4'=p_5=p_H$.

With increase of flight speed there is increased the propeller turbine work

$$L_{\text{тв}}=L_{\text{в}}=118T_4^*\left(1-\frac{1}{\pi_{\text{тв}}^{*0.25}}\right)\eta_{\text{тв}}^*$$

in accordance with increase of the pressure drop on it. The gas temperature before the low-pressure turbine

$$T_4^*=T_3^*-\frac{L_{\text{тк}}}{118}=T_3^*\left[1-\left(1-\frac{1}{\pi_{\text{тВД}}^{*0.25}}\right)\eta_{\text{тВД}}^*\right]$$

remains constant, since we assume that the pressure drop in the high-pressure turbine is kept constant.[1]

Thus, as the flight speed rises, the flow rate of air through the engine is increased, work of the low-pressure turbine rises and the propeller output of the engine is increased. Effective fuel consumption of the engine is decreased. Consequently, as compared to the characteristic of a single-shaft turboprop engine the values of $P_{\text{в}}$ and $N_{\text{в}}$ of a two-shaft turboprop engine are somewhat larger, and accordingly C_e is less thanks to the higher values of $\eta_{\text{в}}$ and $\eta_{\text{тв}}^*$.

Examination of the high-speed characteristics of turboprop engines leads to the conclusion that the propeller output grows with rise of flight speed, and the effective fuel consumption is lowered <u>without limit</u> in the whole range of subsonic and supersonic flight speeds.

Does this signify that the application of a turboprop engine is more profitable, the higher the M_0 number of flight? No, such a conclusion would be absolutely incorrect.

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[1]This is true in those cases when the exit velocities from the primary nozzle box of the low-pressure turbine reach critical values.

Basic efficiency criteria of the turboprop engine are not propeller output, but total thrust, not effective fuel consumption, but specific fuel consumption (referred to 1 kgf of total thrust).

From expressions of total thrust and specific fuel consumption of a turboprop engine

$$P_{\Sigma}=\frac{75N_{\text{в}}\eta_{\text{в}}}{c_0}+R$$

and

$$C_{y\Sigma}=3600\frac{G_{\text{т}}}{P_{\Sigma}}=3600\frac{m_{\text{т}}}{\frac{75L_{\text{в}}\eta_{\text{в}}}{c_0}+\frac{c_5-c_0}{g}}$$

it follows that total thrust of a turboprop engine drops continuously depending on the M_0 number, and specific fuel consumption grows, even in the ideal case when $\eta_{\text{в}}=1$. However, the drop of propeller efficiency at high transonic flight speeds noticeably aggravates the lowering of P_{Σ} and growth of $C_{y\Sigma}$.

If with the same gas generator (i.e., rated values of $G_{\text{в}}$, $\pi_{\text{к}}^{*}$ and T_3^{*}) we analyze the comparative shape of high-speed characteristics of a turboprop ($x_0 = 1$) and turbojet engine ($x = 0$), the conclusion can be made that the very considerab'e advantage of the turboprop engine over the turbojet in developed thrust and economy on a stand (by 4-5 times) is rapidly reduced with rise of M_0 number. Even at transonic M_0 numbers ($M_0 \approx 0.85$-0.90) because of the sharp drop of propeller efficiency (Fig. 5.22) there sets in equalizing of thrusts and specific fuel consumptions of turboprop and turbojet engines (Fig. 5.23).

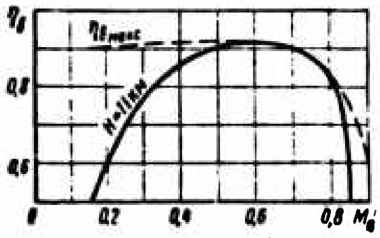

Fig. 5.22. Change of propeller efficiency depending on the M_0 number of flight.

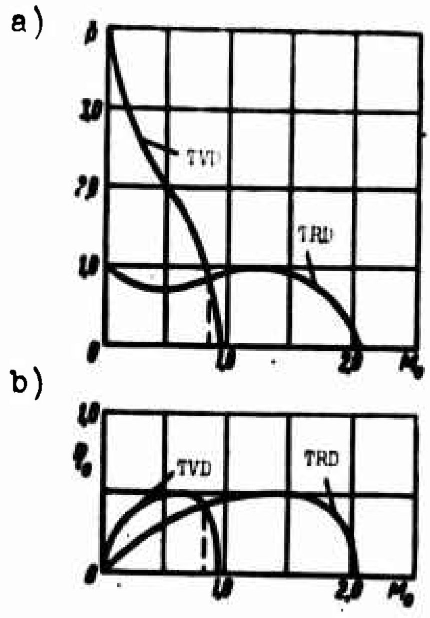

Fig. 5.23. Comparison of the high-speed characteristics of turboprop and turbojet engines: a) thrust; b) overall efficiency.

Thus, the region of expedient use of the turboprop engine with respect to economy is still, as 15-20 years ago, limited to subsonic flight speeds. Attempts at spreading this region to supersonic flight speeds did not give the expected results mainly due to the difficulties of creating supersonic propellers, having sufficiently high efficiency on the stand and at subsonic flight speeds.

§ 3. High-Altitude Characteristics of Turboprop Engines

The dependences of propeller output or total (equivalent) horsepower, and also effective fuel consumption on the flight altitude at a prescribed program of control are called the altitude characteristics of turboprop engines. The altitude characteristic can be obtained in a flying laboratory, and analytically similar to high-speed characteristic.

Altitude Characteristics of Single-Shaft Turboprop Engines

Let us examine the altitude characteristics of a single-shaft turboprop engine with total expansion of gas in the turbine at c_0 = const and with a program of control including the following conditions:

1) n = const;

2) T_3^* = const;

3) f_5 = const and $p_4^* = p_5 = p_H$.

Conditions 1) and 2) are fulfilled by using two regulators (prop and fuel feed), united in the command-fuel unit (KTA).

With increase of the flight altitude due to rise of the overall compression ratio the pressure drop on the turbine is increased and the KTA automatically loads the propeller, keeping the number of engine revolutions and the gas temperature before the turbine constant.

Change of $N_в$ Depending on the Flight Altitude.

With climb to altitude the mass flow rate of air through the engine is continuously decreased, and by the same regularity as for a turbojet engine; specific propeller work increases because of the increase of $\pi_т^*$. Since the fall of airflow rate is determining as compared to the change of $\pi_т^*$, then the propeller output of the turboprop engine also drops with climb to altitude, however, slower than $G_в$ (Fig. 5.24). The drop of $N_в$ at altitudes over 11 km is

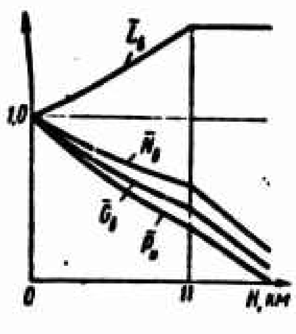

Fig. 5.24. Change of the propeller output of turboprop engines with climb to altitude.

intensified, since in this case the specific work of the propeller keeps its value constant. This is explained by the fact that due

to T_H = const the increase of π_e and π_t^* is ceased.

At altitudes H > 11 km we have

$$N_e \sim p_H. \qquad (5.10)$$

<u>Change of C_e Depending on the Flight Altitude.</u>

With climb to altitude the effective efficiency of the cycle η_e increases, which is caused by the growth of the total compression ratio $\left(\pi = \frac{p_2^*}{p_H}\right)$ and the degree of preheating $\left(\Delta = \frac{T_3^*}{T_H}\right)$. Increase of the effective efficiency leads to a proportional lowering of

$$C_e = \frac{632}{H_u \eta_e},$$

which continues right up to an altitude of 11 km (Fig. 5.25). With further climb the parameters η_e and C_e remain constant due to T_H = const.

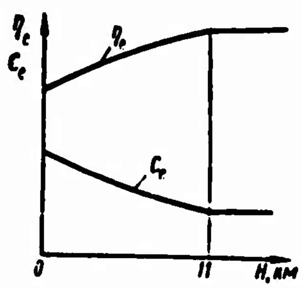

Fig. 5.25. Change of effective fuel consumption of a turboprop engine with climb to altitude.

Analysis of expression C_e by formula $C_e = 3600 \frac{m_T}{N_{ey3}}$ brings us to the same results, since increase of L_e turns out to be the factor dominating the growth of $m_T \sim (T_3^* - T_2^*)$.

High-Altitude Turboprop Engines

With climb to altitude the propeller output of a turboprop engine drops continuously. Therefore, a turboprop engine, similar to the turbojet or piston engines, is an unboosted engine.

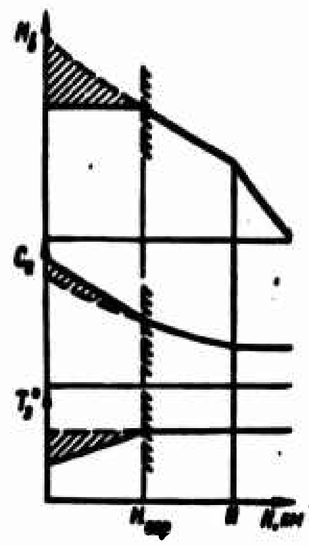

Fig. 5.26. Altitude characteristic of a "high-altitude" turboprop engine.

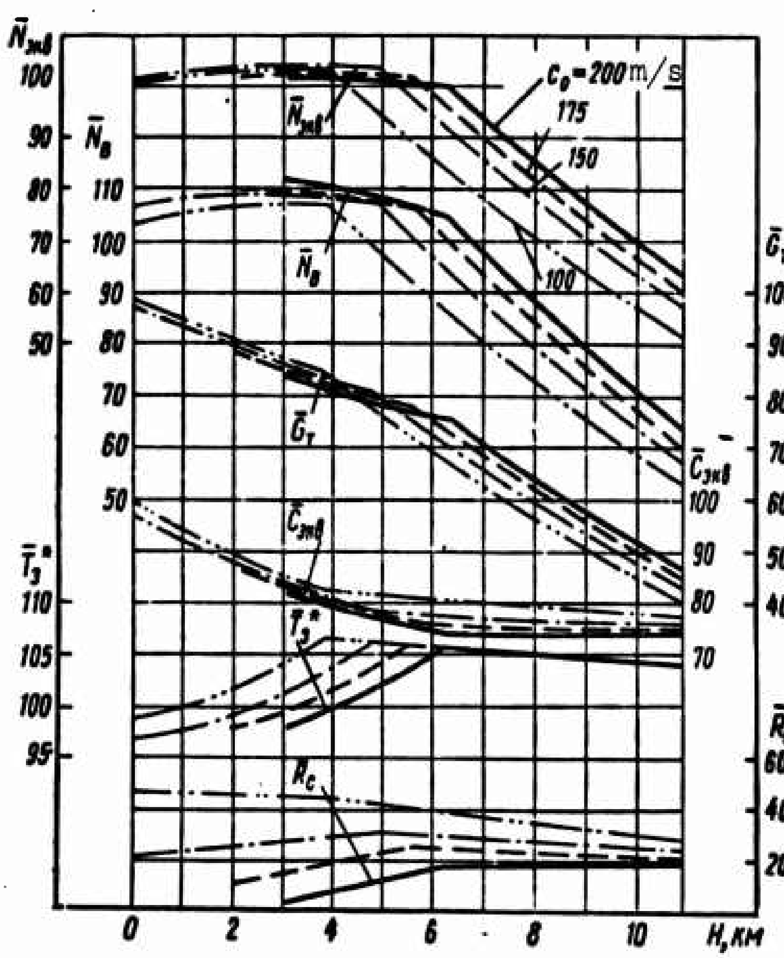

Fig. 5.27. Altitude and high-speed characteristics of a single-shaft turboprop engine.

Recent years have been marked by the intense development of so-called high-altitude turboprop engines, the power of which remains constant with climb to a specific altitude — altitude of limitation. A distinctive feature of these engines consists of the fact that they are calculated for strength not on the ground, but at the mentioned altitude of limitation, i.e., under conditions of lowered density of the medium, and consequently, at lowered values of aerodynamic forces, having an effect on the blades of the compressor and turbine, and also power and circumferential forces, having an effect on the elements of the planetary reduction gear.

Thus, such a turboprop engine, having the necessary safety factor in high-altitude and high-speed conditions, turns out to be relieved as compared to usual turboprop engines, designed for strength at maximum flight speed near the ground.

In proportion to lowering of the flight altitude of an aircraft with turboprop engine the loads transmitted to the main engine joints increase, and it becomes necessary to limit the value of parameter N_e. For this purpose it is necessary to lower the engine revolutions or gas temperature before the turbine (simultaneously relieving the propeller). For such turboprop engines the altitude characteristic has the form depicted in Fig. 5.26. Lowering of temperature T_3^* at altitudes less than the altitude of limitation leads to increase of C_e.

With increase of flight speed the throttling T_3^* is made deeper. The altitude of limitation increases accordingly.

On Fig. 5.27 there are given altitude and high-speed characteristics of a high-altitude single-shaft turboprop engine.

CHAPTER VI

UNSTABLE OPERATION OF GAS-TURBINE AIRCRAFT ENGINES AND MEASURES FOR PREVENTING

The operation of gas-turbine engines in a number of cases is hampered by the appearance of unstable operating conditions.

Unstable operation of gas-turbine engines is usually manifested in pulsations of air (gas) flow in the gas-air duct of the engine, which leads to a drop of thrust, increase of the gas temperature before the turbine, vibrations and even failure of separate elements of the gas-turbine engine structure.

The operation of gas-turbine engines at the appearance of unstable conditions is impermissible.

In literature unstable engine operation is usually called surge. The term "surge" signifies fluctuating, unstable operation of the compressor. Unstable operation of a turbojet engine is a more widely spread phenomenon than unstable operation of the compressor, one of its elements.

Causes of the appearance of unstable operation of gas-turbine engines can be compressor surge, sharp movement of the engine control lever, and also special burning conditions of the fuel-air mixture in combustion chambers of gas-turbine engines — resonance burning.

Unstable operation of gas-turbine engines most frequently appears during engine starting (especially in winter), under transient

conditions (at sharp movement of RVT), at maximum revolutions (mainly in winter and in flight at high altitudes). For the competent operation of gas-turbine engines the engineering and technical staff of subdivisions of civil aviation should be well acquainted with the physical essence of unstable operation, criteria of its appearance, effect on engine operation and harmful consequences, and also measures of combatting it.

§ 1. Physical Essence of Unstable Compressor Operation

Unstable compressor operation appears in the form of intermittent and severe fluctuations[1] of air flow – oscillations of pressure, speed and flow rate. Average pressures, developed by the compressor, usually drop, and the compressor inlet temperature considerably increases. At times the compressor as if "floods," ejecting the masses of air in the opposite direction, to the inlet. Surging is accompanied by severe backfires and knocks.

At present it is established that surging is connected with intermittent burbles, appearing mainly on the convex surface (back) of blade profiles during flow around the compressor grids. At constant compressor revolutions (and consequently, at constant peripheral velocity of the blades) the decrease of flow rate leads to decrease of the axial component of the flow velocity at the inlet to the given grid. Consequently, the relative velocity during flow around the profile in the grid changes its direction, the incident flow angle, being increased, becomes greater than critical, due to which there appears separation of flow from the blade back.

Figure 6.1 shows a diagram of stalled flow around the airfoil lattice of the rotor wheel of an axial-flow compressor. Vortexes appearing during flow separation are unstable and are prone to self-growth. The forming vortex sheet, spreading in the vane

[1]Fluctuations of flow during surge carry a self-oscillation character, i.e., depend only on the properties, elasticity of the oscillating system itself, for example, capacity of air and gas duct, but do not depend on disturbing force, for example, the number of revolutions of the compressor.

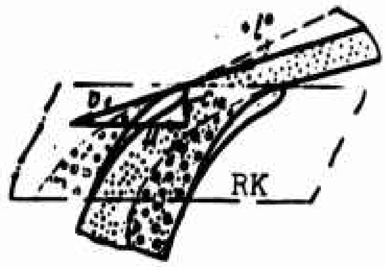

Fig. 6.1.

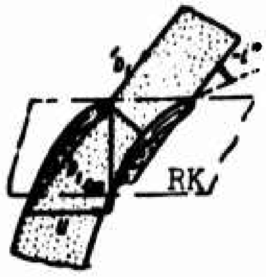

Fig. 6.2

Fig. 6.1. Diagram of stalled flow around the airfoil lattice of axial-flow compressor (surging).

Fig. 6.2. Flow around the airfoil lattice of an axial-flow compressor under turbine conditions.

channel,[1] reduces the cross section of flow, as a result of which the airflow rate is reduced still more. There comes a point when the vortexes completely fill the vane channels, and supply of air by the compressor in this case is ceased (the airflow rate is equal to zero). In the following instant the vortex sheet is washed away, and in this case ejection of air to the compressor inlet is possible. Repeated and multiple compression of the same portion of air in the compressor during surge leads to increase of the air temperature at the compressor inlet (multiple feed of energy to the same mass of air).

With increase of the airflow rate (negative profile angles of attack) flow separation occurs from the concave side of the blade. Vortexes forming in this case are pressed to the profile by the basic flow and have a stable character (Fig. 6.2). Turbine conditions set in. The given explanation of the physical essence of phenomena leading to surge is in accordance with the shape of characteristic (Fig. 6.3): surge corresponds to its left branch, turbine and choking conditions – to the extreme end of the right branch.

[1]Under the action of centrifugal forces of inertia.

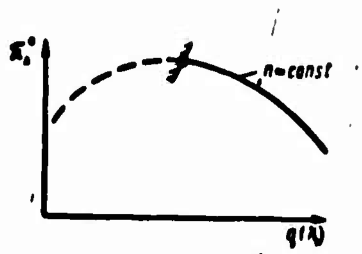

Fig. 6.3. Explanation of unstable compressor operation.

Experimental Investigation of Surge

Oscillations of flow (pressure, flow rate) during compressor surge can be recorded by an oscillograph (loop or cathode) with the aid of receiving devices – throttle valves, equipped with sensing devices, or wire anemometers.

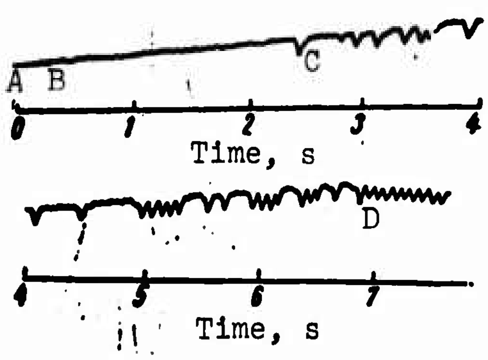

Fig. 6.4. Typical oscillogram of total air pressure at the compressor outlet.

Figure 6.4 shows a typical oscillogram of the total air pressure at centrifugal compressor outlet. With throttle covering at point A there start to appear insignificant isolated oscillations of pressure. As the flow rate decreases the oscillations are amplified (point B) and become periodic. With further decrease of flow rate the oscillations are amplified still more and become intermittent (point C). Then the oscillations again become periodic (point D).

Experimental investigations of surging in axial-flow compressors during the last 10-15 years permitted revealing some new regularities. Two different types of surging were revealed. The first, sometimes called progressive[1] surge, is characterized by the formation of

[1]The names "progressive" and "total" surge are conditional to a considerable degree.

a vortex zone, spreading only to part of the compressor blade along its height and, consequently, encompassing only part of the annular section of flow. The second, called total surge, spreads to the entire annular section of flow from the root of the blade to its tip.

Progressive surge appears in stages with small hub-tip ratio ("long" blades), when

$$\bar{d} = \frac{d_B}{d_H} < 0,6,$$

where d_B — hub diameter; d_H — external diameter of blade.

It mainly encompasses the region of peripheral sections. The nature of this type of unstable operation is caused by the fact that with large blade length under the action of centrifugal forces of inertia there occurs intense overflow of the boundary layer from the root of the blade to the tip and formation of peripheral revolving areas of separation.

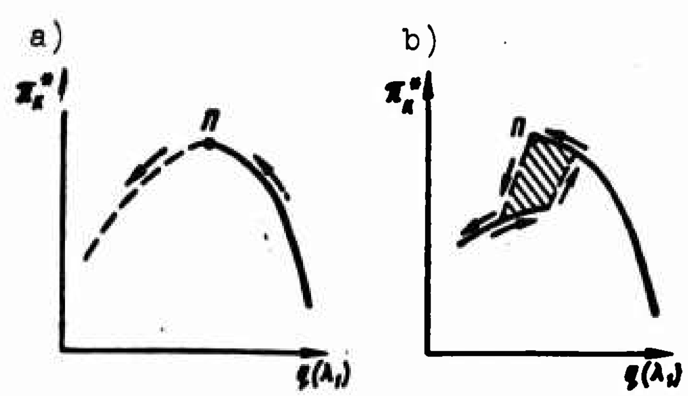

Fig. 6.5. Characteristic of compressor in the presence of progressive surge (a) and total surge (b).

Progressive surge is characterized by a smooth and continuous drop of pressure as the flow rate decreases (Fig. 6.5).

Total surge sets in at a relatively large hub-tip ratio, when the angles of attack of flow along the blade are hardly changed.

It can generate from progressive with further considerable decrease of the flow rate, when flow separations spread to the entire span of the blade.

A peculiarity of total surge during its generation is the sharp drop of compressor outlet pressure (reaching up to 1/3 of the initial value of pressure) and the presence of a discontinuous characteristic.

Rotary Stall

The experimental investigation of surge in recent years revealed the existence of a special type of unstable operation of an axial-flow compressor — so-called rotary stall. The separation area, encompassing the group of blades, travelling along the circumference of the rotor wheel, is called rotary stall. Rotary stall usually appears at the periphery of the blades, and then spreads to their hub. The speed of rotation of separation is proportional to the peripheral velocity of the blades. In each stage of the compressor there can exist several rotary stall zones.

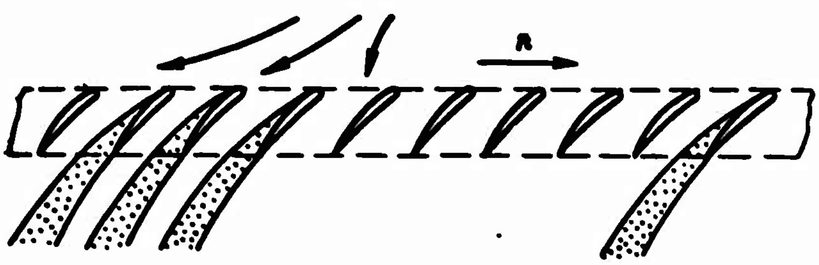

Fig. 6.6. Diagram of appearance of rotary stall.

The appearance of rotary stall can be explained in the following way. Let us assume that in the group of blades there appeared local flow separation. Since in this case there occurs partial or total covering of the vane channels by vortices, this leads to spreading of the flow before the grid (Fig. 6.6). As a result the angles of

attack of blades located to the right of the separation will be decreased, which will improve their streamlining and prevent the appearance of flow separations. Conversely, angles of attack of blades located to the left of the separation will increase; this will impair their streamlining and lead to flow separation. The stall region will start to move from right to left, opposite the rotation of the rotor.

Rotary stall impairs compressor characteristics, but on the whole the compressor can operate stably.

Usually the quantity of the blades enveloped by rotary stall is kept constant.

§ 2. Mutual Location of Surge Lines and Lines of Operating Conditions of the Compressor and Turbine. The Concept of a Margin of Stability with Respect to Surge

The totality of points of the beginning of surge at various revolutions will form the so-called surge line (Fig. 6.7). With decrease of numbers of revolutions the surge line is displaced in the direction of smaller flow rates.

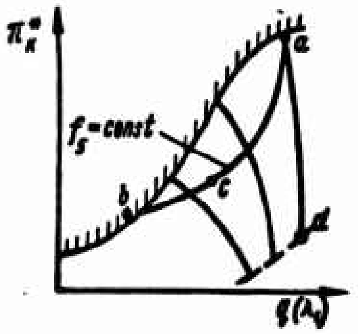

Fig. 6.7. Compressor characteristic.

The totality of performance points in compressors in turbojet engines will form characteristic compressor curve a-c-b. Its shape and course on the characteristic depend on conditions of the joint work of the compressor and turbine, and also the program of engine control.

To ensure normal gas-turbine engine operation in the entire range of numbers of revolutions, speeds and altitudes of flight the characteristic curve of the compressor should pass to the right of the surge line.

Equation of line of operating conditions of the gas-turbine engine on the compressor characteristic can be obtained by using conditions of the joint work of compressor and turbine in a turbojet engine or the compressor, turbine and propeller in a turboprop engine. We obtained such LRR equations for turbojet, ducted fan and turboprop engines in Chapters II-V.

In reference to the compressor of a turbojet engine with fixed geometry (f_5 = const) the LRR equation has the form

$$\frac{\pi_K^*}{\sqrt{\frac{\pi_K^{*0,286}-1}{\eta_K^*}}}=Cq(\lambda_1).$$

This equation is depicted on the compressor characteristic by a parabolic curve, the course of which relative to the surge line depends on the value of $\pi_{K(p)}^*$.

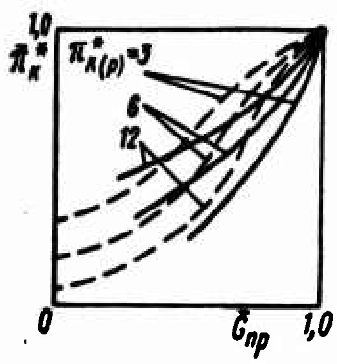

Fig. 6.8. Effect of the compressor compression ratio on the combined passage of lines of operating conditions and the surge line.

On Fig. 6.8 in relative coordinates $\bar{\pi}_K^*=f(\bar{G}_{np})$ there is shown the effect of the calculated compression ratio of the compressor on the slant of curves of the surge line and lines of operating conditions. With increase of $\pi_{K(p)}^*$ the surge line is steeper, and the line of operating conditions – flatter. Thus, the danger of

appearance of compressor surge at lowered revolutions increases with rise of the compression ratio of the compressor. Conversely, at small values of $\pi^*_{\kappa(p)}$ the danger of appearance of surge at raised revolutions is increased.

Concept of Surge Margin of Stability (Fig. 6.9)

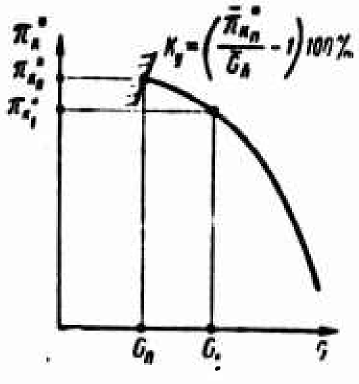

Fig. 6.9. Determination of the surge margin of stability of the compressor.

<u>Stability margin of the compressor with respect to surge</u> is the name for expression

$$k_y=\left(\frac{\bar{\pi}^*_{\kappa_n}}{\bar{G}_n}-1\right)100\%, \qquad (6.1)$$

where $\left.\begin{array}{l}\bar{\pi}^*_{\kappa_n}=\dfrac{\pi^*_{\kappa_n}}{\pi^*_{\kappa_1}}\\ \bar{G}_n=\dfrac{G_1}{G_n}\end{array}\right\}$ – relative values of the compression ratio and airflow rate on the surge line.

With vertical characteristic of compressor $(\bar{G}_n=1)$

$$k_{y(\pi)}=\frac{\Delta\pi^*_\kappa}{\pi^*_{\kappa_1}}=\frac{\pi^*_{\kappa_n}-\pi^*_{\kappa_1}}{\pi^*_{\kappa_1}}\cdot 100\%$$ – relative increase of compression ratio, sufficient for the appearance of surging.

With horizontal characteristic of compressor $(\bar{\pi}^*_\kappa=1)$

$$k_{y(G)}=\frac{\Delta G}{G_1}=\frac{G_1-G_n}{G_1}\cdot 100\%$$ – relative lowering of the airflow rate, sufficient for the appearance of surging.

The stability margin of axial-flow compressors of gas-turbine aircraft engines under the rated conditions should be not less than 12-17%.

Surge in High-Pressure Axial-Flow Compressor

A peculiarity of operation of the multistage high-pressure axial-flow compressor is "mismatch" or "divergence" of operation of the extreme (i.e., the first and the last) stages under partial load conditions.

Let us assume that at rated operating conditions (for example, at nominal revolutions) all the compressor stages work stably, without separations of flow, with the computer values of compression ratios and efficiency. If the compressor does not have special control elements (for example, throttle valves at the inlet and exit), then any change of its operating conditions, determined by the number of revolutions, leads to a change of the blade angles of attack of primarily the extreme stages and to a smaller degree the middle stages. As the revolutions decrease the angles of attack on the first stages increase, and decrease on the last. As a result on the first stages of the compressor there appears separation of flow from the blade backs and as a result of this – surge; on the last stages there appears so-called turbine conditions, which are characterized by a sharp drop of compression ratio, and also "blocking" conditions.

Mismatch of the work of extreme stages turns out to be greater, the lower the revolutions and the higher the value of compression ratio of the compressor at initial rated conditions.

With increase of revolutions above nominal the mismatch of work of the extreme stages is changed – now there even appears surge on the last stages; on the first stages with the appearance of sonic and supersonic relative flow velocities there appear blocking conditions. Figure 6.10 shows a diagram of flow around the blades of the first (1), middle (m) and last (z) stages of an axial-flow

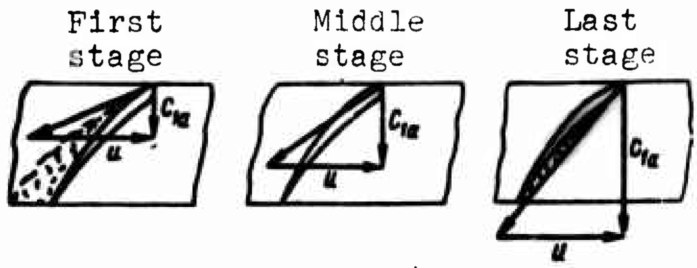

Fig. 6.10. Diagram of flow around blades of the first, middle and last stages of an axial-flow compressor at lowered revolutions.

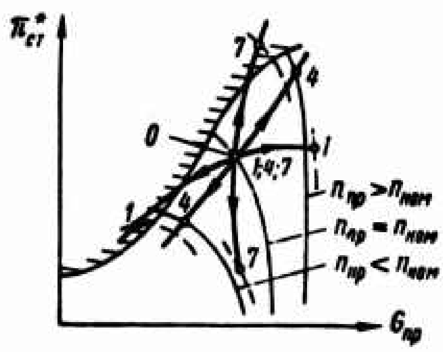

Fig. 6.11. Mismatch of the work of extreme stages of an axial-flow compressor at low and high revolutions.

compressor at reduced revolutions, and Fig. 6.11 shows the combined characteristics of the first, middle and extreme stages of a compressor with the lines of operating conditions of these stages plotted.

§ 3. Effect of Operating Conditions on the Margin of Stable Compressor Operation

Starting the Turbojet Engine

Compressor surge is possible when starting the turbojet engine because of a short-duration increase of gas temperature before the turbine ("sudden temporary rise" of temperature). This can occur during a too early cutoff of the starter (with respect to the revolutions), when excess turbine power for spinning up the rotor turns out to be insufficient, and also with sudden opening of the fuel cock.

The starter can be cut off earlier (regardless of the power ratio of the turbine and compressor), when the duration of its action is limited by a timer. Early cutoffs of the starter are observed in winter, when the viscosity of oil for lubrication of rotor bearings is greater than usual and, furthermore, greater than the usual mass flow rate of air through the compressor. Increase of power, required for engine rotor spinup in this case, is one of the causes of "sudden temporary rise" of temperature before the turbine and compressor surge when starting.

Sudden opening of the fuel cock causes excessive enrichment of the mixture, as a result of which the flame jet before the turbine is extended, combustion turns out to be incomplete, and accumulation of unburned fuel in the jet nozzle is possible. Combustion of this fuel can carry an oscillatory character ("resonance burning") in the jet nozzle and usually leads to compressor surge (so-called "rumbling" of the engine).

Maximum Revolutions

At maximum revolutions there is also possible the appearance of surge, especially for those engines with small values of $\pi_{кр}$, for which the line of operating conditions and the surge converge. In an axial-flow compressor at maximum revolutions the stability margin of the last stages is reduced in connection with the fact that the angles of attack sharply increase on them.

Increase of Flight Speed (Fig. 6.7)

With transition to supersonic flight speeds (at constant nominal or maximum revolutions) the given engine revolutions

$$n_{пр}=n\sqrt{\frac{288}{T_H^*}}$$

are sharply reduced, since in this case the temperature of air at the compressor inlet is considerably increased, which leads to

displacement of the performance point of the turbojet engine along the line of combined conditions (a-c-b) and to decrease of the margin of stability. In this case surge appears on the first stages of the compressor.

Increase of Flight Altitude (Fig. 6.7)

With climb to altitude (when n = const) the given rotor revolutions increase (since the air temperature at the compressor inlet is decreased) and the performance point shifts along line of combined conditions (a-c-b), approaching the surge line. At high altitudes the appearance of surge is possible in the last stages of the axial-flow compressor.

Effect of External Atmospheric Conditions

Change of the external pressure does not affect the performance point of the compressor and does not change its margin of stability.

Change of ambient temperature leads to change of the given revolutions, and consequently, to a shift of the performance point of a turbojet engine on the compressor characteristic. Considerable lowering of the ambient temperature at maximum revolutions may cause surge in the last stages of an axial-flow compressor.

Unstable Operation at Transient Conditions

During operation under transient conditions, in particular during the transition from idle to nominal revolutions, a sharp movement of the RUD can lead to an instantaneous "temporary rise" of the temperature before the turbine above its maximum value and can cause compressor surge.

For preventing surge at transient conditions the contemporary turbojet engines are equipped, as a rule, with an automatic device or control valve for acceleration, which ensures the necessary ratio between the airflow rate and fuel consumption.

§ 4. Measures for Preventing Compressor Surge

Measures for preventing compressor surge can be subdivided into operational and structural.

Operational measures are those which are used to prevent surge, but upon its appearance, to remove it.

The possibility of appearance of surge is prevented by assuring the correct fuel metering with respect to revolutions both during starting and at transient operating conditions of the engine (application of automatic acceleration controls),[1] and also the required engine rotor spinup by the starter.

In cases of the application of an electric starter, the above-stated considerably depends on the degree of charge of the batteries. In connection with this, instructions for engine maintenance prohibit the application of batteries, whose voltage is lower than set limits, for starting

With the appearance of instability of engine operation it is necessary to decrease the revolutions of the engine rotor by the RUD until the disappearance of surge, to lower the altitude of flight, or to increase the flight speed. Any of these measures lowers the given revolutions and shifts the performance point to the stable region of the characteristic.

Structural measures involve:

1) shifting the surge line to the region of smaller flow rates with prescribed line of joint conditions of the compressor and turbine. This is done by application of rotary blades of the stator or rotor at the axial-flow compressor inlet and exit, by special selection of blade angles of the rotor wheel of the axial-flow compressor, etc;

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[1]See Chapter IX.

2) shifting the line of joint conditions of the compressor and turbine to the region of high flow rates at a prescribed surge line of the compressor. This is done by application of a variable-area jet nozzle and nozzle box of the turbine;

3) providing matched and surge-free operation of all the stages of the high-pressure axial-flow compressor. For this purpose there are applied blow-bys from the intermediate compressor stage into the atmosphere, rotary stator blades, and also two-shaft axial-flow compressors.

Application of Rotary Stator Blades at the Compressor Inlet and Exit

The principle of elimination of surge in any compressor stage with the help of rotary stator blades consists of the creation of a preliminary twist of the flow at the rotor wheel inlet in the direction of rotation of the rotor wheel (Fig. 6.12).

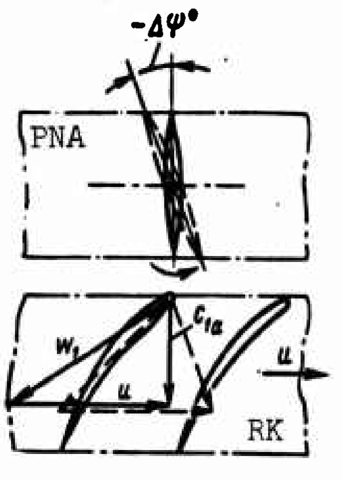

Fig. 6.12. Diagram of the stage of an axial-flow compressor with rotary stator blades at the inlet [RK = rotor wheel].

At constant values of circumferential u and axial c_{1a} velocities in the rotor wheel, the creation of a preliminary twist of flow in the direction of rotation of the rotor changes the direction of relative velocity w_1 at the rotor wheel inlet so that the angles of attack of the rotor blades would decrease. In this case the normal flow around the blades is restored.

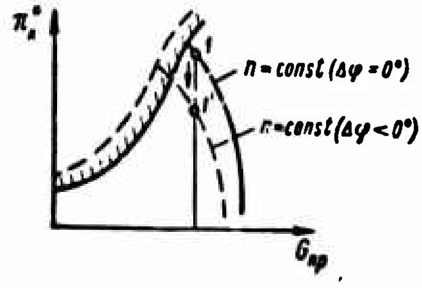

Fig. 6.13. Effect of turning the stator blades on the compressor characteristic.

The flow is pretwisted by turning the stator blades to angle $-\Delta\varphi^{\circ}$, counterclockwise.

Turning the stator blades changes the compressor characteristic, displacing lines n = const and the surge to the region of lowered airflow rates (Fig. 6.13).

Elimination of blocking conditions or turbine conditions in the stage can be provided by turning the stator blades in the direction opposite the rotation of the rotor (creation of a preliminary twist of flow). In this case the angles of attack on the rotor blades increase, and normal flow around the blades is restored.

The best solution of the problem of providing stable compressor operation would be to make a number or even all the compressor stators rotary with variable angles of rotation on the different stages. In certain cases this is done, although it essentially complicates the compressor construction. Thus, for example, the compressor of a J-79 turbojet engine has six rotary stators out of seventeen.

Most often we resort to the application of a rotary stator at the compressor inlet.

Application of Blow-By

Bypassing the air from the intermediate stages of the compressor to the atmosphere gives the possibility of coordinating the work of all the stages of a high-pressure compressor. Constructively this is carried out so: in the compressor housing over the entire

external circumference of the grid of the given stage there are made bypass ports, which are closed by a steel strip. When necessary the strip is removed from the ports and passes the air into the atmosphere. For bypass of air there are also applied valves with a controllable butterfly.

The bypass mechanism is controlled automatically and manually.

For high-pressure axial-flow compressors the surge always appears at low and medium revolutions (because of peculiarities of the mutual course of the line of joint conditions and surge line; see Fig. 6.7), therefore, their bypass strip during starting is in the open position up to specific revolutions, at which the danger of appearance of surge does not appear.

Let us examine the principle of action of the blow-by system (Fig. 6.14).

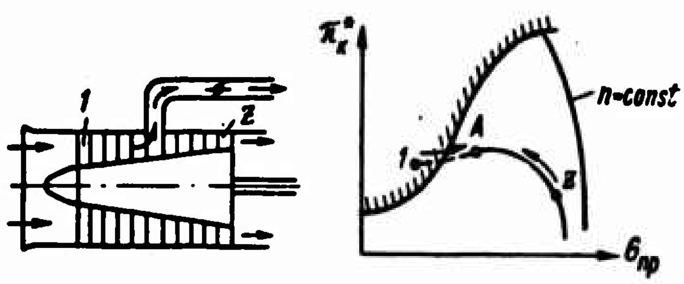

Fig. 6.14. Coordination of work of the extreme compressor stages by blow-by.

Let us assume that air is discharged from the middle stage of the compressor. The blow-by system hydraulically constitutes an additional channel with a throttle, connected in parallels to the basic channel.

With opening of the bypass throttle due to decrease of drag of the line the flow rate of air through the compressor stages, set before the bypass ports, increases. Consequently, the axial velocities of the airflow increase, and angles of attack on the first

stages are decreased (see Fig. 6.2). The stability margin of surge is increased in this case, reference point 1 (see Fig. 6.14) shifts along the characteristic in the direction of greater flow rates.

The airflow rate through the stages, set past the bypass ports, is decreased, since a considerable part of it flows out through the bypass ports with less hydraulic losses. This leads to the fact that in the last stages the axial flow velocities of air are reduced, angles of attack increase, and blades emerge from turbine conditions.

Decrease of axial flow velocities in the last stages (with open bypass strip) is also explained by the fact that increase of head of the first compressor stages when bringing them from surge leads to an increase of density, and consequently, to a decrease of the volumetric flow rate through the last stage. Thus, the operating point of the last stage (z) shifts along characteristic n_{np}=const in the direction of smaller flow rates.

Application of Split Compressors

The transition to a split compressor of a turbojet engine (see Fig. 1.2), rotated by two separate turbines with a different number of revolutions for each shaft, is a reliable and effective means of providing of stable and coordinated operation of all the compressor stages at different operating conditions on the ground and in flight.

The principle of coordination of the operation of the extreme compressor stages by changing to a split compressor is easy to explain, proceeding from redistribution of the axial velocities along the compressor at reduced or increased given revolutions (or from consideration of the change of blade angles of attack along the compressor duct).

As was shown earlier, at reduced revolutions the blade angles of attack are increased, on the last stages they are decreased. So that the blade angles of attack would not deviate to any side from

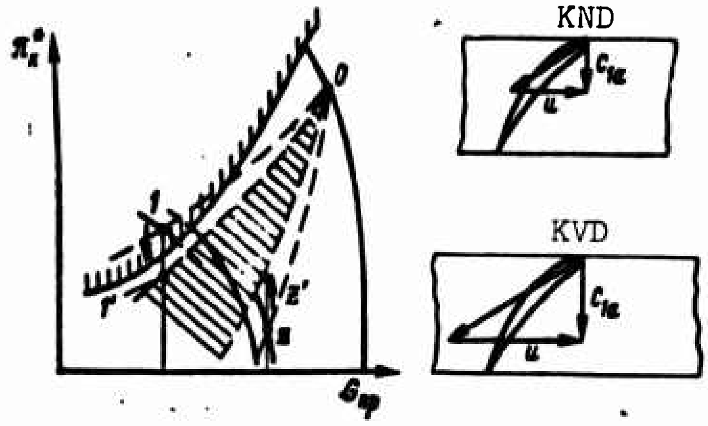

Fig. 6.15. Coordination of work of high-pressure compressor stages by changing their revolutions (application of ducted-fan engine).

their computed value, in accordance with the change of axial velocity at the new conditions it is necessary to change the peripheral velocity. For this on the first stage of the compressor it is necessary to additionally decrease the peripheral velocity, and on the last stage, to increase it somewhat. Thus, at lowered revolutions for coordination of operation of the extreme stages the rotor wheels of the first stages must revolve at less revolutions than the last (Fig. 6.15).

At very high given revolutions, when angles of attack of the first stages are decreased, and the last are increased, it is necessary that the number of revolutions of the rotor wheels of the first stages be greater than the last stages.

Theoretically the best solution of the problem would be to provide a variable number of revolutions of each stage. However, it is very difficult to do this structurally, and is not even necessary. It is sufficient to divide a multistage compressor into two rotors (two stages) so that the degree of increase of pressure of each would not exceed 3.5-4.5, with which all the stages still operate sufficiently coordinated. A two rotor compressor can provide stable and coordinated operation of the stages of an axial-flow

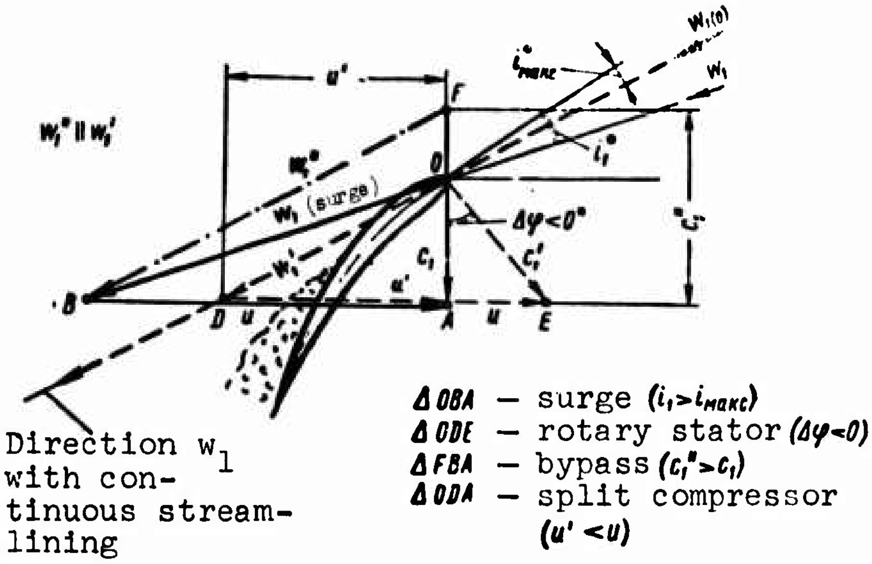

Fig. 6.16. Schematic representation of the basic principles of preventing surge of the stage of an axial-flow compressor.

compressor with total degree of pressure increase 12-20.[1] As compared to an initial single-shaft compressor at reduced revolutions with the same value of flow rate and compression ratio the first stage of a split compressor has a lower number of revolutions, and the second – a higher number of revolutions.

A two-shaft turbojet engine possesses the property of self-coordination of operating conditions of compressor stages. This process occurs due to the mutual "slip" of rotors (change of the relationship between the revolutions of high- and low-pressure compressors under partial load conditions). Explanation of the given phenomenon is given in examining the throttle, altitude and high-speed characteristics of turbojet engines.

On Fig. 6.16 there are schematically represented the methods of eliminating the generated surge of the stage of an axial-flow compressor with the aid of a rotary compressor stator (introduction of a preliminary twist of flow at the rotor wheel inlet), blow-by

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[1]When $\pi_к^* >$ 20-25 the compressor is three-shaft (ducted-fan engine RB-178, "Trent" and others).

(increase of axial flow rate), transition to a two-shaft scheme (lowering of the peripheral velocity of rotation of the rotor).

CHAPTER VII

THE EFFECT OF OPERATIONAL FACTORS ON GAS-TURBINE ENGINE OPERATION. OPERATIONAL RELIABILITY OF THE ENGINE

§1. Factors Affecting Operating Conditions and Parameters of Gas-Turbine Engines

The operating conditions and parameters of gas-turbine engines are essentially affected by the operational factors mentioned below.

External and Flight Conditions

External and flight conditions include the state of the external atmosphere (pressure and temperature of air, humidity of the atmosphere), and also the speed and altitude of flight. Their effect on engine operation appears through a change of the total parameters of air p_H^* and T_H^*..

It was noted above that a decrease of T_H in winter leads to substantial augmentation of turbojet engine thrust or turboprop engine power. Accordingly there increase the moments and forces in structural elements of the engine, and consequently also stresses. Thus, starting from certain values of T_H, it is necessary from strength considerations to introduce engine thrust limitation. This is done, for example, by maintaining a constant fuel consumption per second

(hour) and by a corresponding lowering of the revolutions and gas temperature before the turbine. At this point, starting with a certain limitation temperature ($T_H < T_{огр}$), thrust (power) of the gas-turbine engine is kept constant or is hardly changed.

At air temperatures higher than the limitation temperature ($T_H > T_{огр}$), thrust drops. However, a considerable drop of thrust in operation is very disadvantageous. It hampers takeoff, requires the introduction of special GTD boosting either by the number of revolutions (which lowers the engine life), by the injection of a water-methane mixture into the compressor inlet, by afterburning of fuel past the turbine. These measures also have specific limitations. Usually it is important to keep thrust constant up to $t_H = 30-40°C$, and then to allow it to decrease.

Figure 7.1 shows the dependence of thrust of the ducted-fan engine "Spey" on the change of external temperature without the injection of a water-methane mixture into the compressor and with it. Injection of water-methane mixture permits keeping the thrust constant up to t_H = 35°C and to provide a 9% increase of thrust when t_H > 35°C.

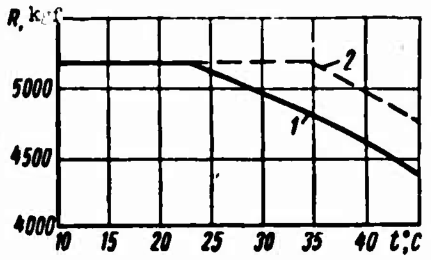


Fig. 7.1. Dependency of DTRD "Spey" thrust on the ambient temperature: 1 – without injection of water-methane mixture; 2 – with injection of mixture.

For the turboprop engine "Tyne" the injection of water-methane mixture give up to 28% increase of effective power.

Figure 7.2 shows the effect of ambient temperature on thrust of the turbojet engine "Avon." At temperatures below +15°C the engine thrust increases somewhat.

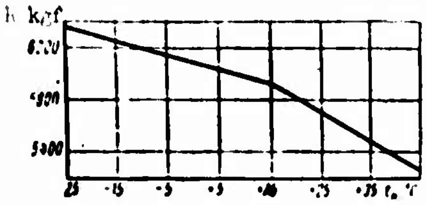


Fig. 7.2. Effect of ambient temperature on the thrust of turbojet engine "Avon."

<u>By humidity of the atmosphere</u> we mean the content of water vapors in it. Since atmospheric humidity is continuously changed from minimum (dry air) to maximum (100-percent humidity, when the atmosphere contains), saturated water vapor it is important to know how it affects the operation and basic indices of an aircraft engine.

The presence of water vapors in air is conveniently estimated by <u>specific or relative humidity</u> q, meaning by it the quantity of water (water vapors) in grams, contained in 1 kg of air. The value of relative humidity is determined basically by the air temperature, and also the pressure of atmosphere.

Numerous engine tests show that the humidity noticeably affects the operation of gas turbine engines. The physical essence of the effect of atmospheric humidity on gas-turbine engine parameters is expressed through change of the gas constant of air, which is increased with growth of q (R_{H_2O} = 47 kgf-m/kgf deg as opposed to $R_в$ = 29.3 kgf-m/kgf deg). This circumstance leads to increase of the specific heat of air, and consequently, to increase of the efficiency of gas and to growth of effective work of the thermodynamic cycle (at prescirbed value of working process parameters). As a result the gas exit velocity from the engine (in proportion to $\sqrt{R}$) and the specific thrust of turbojet engines increase.

On the other hand, the presence of water vapors in air lowers its specific gravity, which ensues from formula

$$\gamma = \frac{p}{RT} \sim \frac{1}{R}.$$

Thus, the mass flow rate of air through the engine drops; it drops more intensively than the specific thrust rises. Consequently,

increase of humidity leads to a drop of turbojet engine thrust

Increase of the speed of sound in humid atmosphere leads to increase of equilibrium numbers of engine revolutions (with preservation of similar turbocompressor conditions).

Figure 7.3 shows the effect of humidity on the basic indices of the turbojet engine "Avon." Humidity of the engine is changed from usual q = 0.01 to maximum q = 0.065.

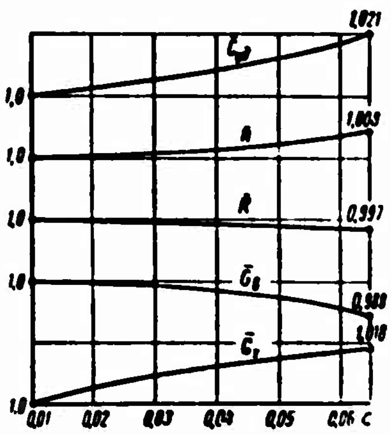


Fig. 7.3. Effect of atmospheric humidity on the basic parameters of turbojet engine "Avon."

It was established experimentally that because of increase of the humidity on a hot day the turbojet engine thrust drops 0.3-0.5%, specific fuel consumption rises 2.1-2.6%, and fuel consumption per hour is increased 1.8-2.1%.

Factors Leading to Additional Gas-Dynamic and Hydraulic Losses in the Gas-Air Duct

Wear of the surface of working elements of the engine, chipping, nicks, appearing because of the presence of dust and other particles in the sucked-in atmospheric air, the appearance of icing at the engine intake can cause additional gas-dynamic and hydraulic losses in different elements of the engine: its intake, compressor, combustion chamber, turbine, afterburner and jet nozzle. When operating the engine at very high altitudes there occurs a sharp lowering of Reynolds numbers and due to this a drop of compressor and turbine efficiency.

Increase of lossses in the gas-air duct, as a rule, disturbs the operating conditions of the engine and impairs its economy.

It is necessary to bear in mind that increase of hydraulic losses in the turbojet engine elements is almost always accompanied by intensification of irregularity of temperature and velocity fields, which additionally impairs the engine operating conditions: the engine is overheated, stall and oscillatory operating conditions appear, vibrations appear in the engine joints.

Factors Leading to Impairment of Carburetion in Combustion Chamber and Afterburner

In a number of cases of operation carburetion deteriorates in the combustion chamber and afterburner, as a result of which the fuel combustion efficiency is lowered and the specific and fueld hourly consumption increase.

Causes leading to the impairment of carburetion can be: pressure drop in the combustion chamber at high altitudes, carbon formation in the fuel nozzles and their clogging, maladjustment of fuel pump, etc.

Maladjustment of the Engine in the Process of Operation or Repair

In the process of operation and repair of the engine, disturbances are possible in the control systems of its units, mechanisms and automatic machines. These disturbances, if they are not revealed and are not eliminated at the proper time, can lead to engine failures and damage. A typical example of maladjustment of the engine is malfunction of the variable-area jet nozzle control mechanism at the moment of fuel feed to the afterburner of a turbojet engine. Because of unopening of the jet nozzle there set in impermissible "sudden rise" of temperatures in the basic cimbustion chamber, compressor surge and other dangerous phenomena. Incorrect selection of the jet nozzle exit section of a turbojet engine during repair leads to analogous results.

Another example of disturbance of the engine adjustment is incorrect installation of the compressor stator during repair, which also can lead to decrease of surge margin, undesirable increase of gas temperature before the turbine, change of thrust and fuel consumption of the engine.

Maladjustment of the fuel system of the engine which disturbs its normal operation, is especially dangerous.

* *
*

The effect of separate operational factors of turbojet engine work when n = const is given in Table 3.

Change of external pressure does not affect the operating conditions of the turbocompressor, since it causes a proportional change of pressure over the entire gas-air duct of the turbojet engine without changing the temperature fields. As a result the exit velocity from the engine and specific fuel consumption keep constant values. The mass flow rate of air through the engine is changed, which turns out to be greater, the higher the external pressure. Total thrust of the turbojet engine is changed in proportion to parameter $G_в$

Change of external temperature displaces the performance point of the compressor along the IRR (when n = const), substantially changes the air flow rate and specific parameters of the engine and changes total thrust especially intensively.

The effect of losses in the gas-air duct on engine operation has some peculiarities. Thus, for example, increase of losses in different elements of the expansion duct (combustion chamber, turbine, afterburner, jet nozzle) leads to the same consequence: sudden temporary rise of temperature before the turbine, displacement of the line of operating conditions to the surge line, impairment of the economy of engine operation. Engine thrust can even be increased in this case.

Table 3. The effect of operational factors on the operating conditions and parameters of turbojet engines (TRDF) when n = const.

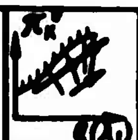
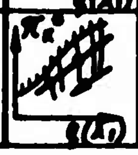
Operational factor	Designation	Change of TK conditions	Change of TRD parameters							Note
			T_3^*	R_{yd}	G_B	R	m_T	C_{yd}	G_T	
Ambient atmospheric conditions										
Ambient pressure	p_0 ↓	Similarity of TK is preserved	const	const	↓	↓	const	const	↓	
Ambient temperture	T_0 ↑		↑	↓	↓	↓↓	↓		↓	$\pi_K^*>10$ L_K↑
Altitude and high-speed conditions										
Increase of flight altitude	H ↑		~const	↑	↓	↓	↑	↓	↓↓	$T_3^* \sim L_K$
Increase of flight speed	c_0 ↑		~const	↓	↑	↓	↓	↑	↑↑	
Engine adjustment										
During boosting the jet nozzle is not opened	T_Φ^* ↑ f_5 = const		↑	↑	const	↑	↑	↑	↑	T^*↑ Surge margin is decreased
Jet nozzle is incorrectly selected during repair	f_5 ↑		↓	↓	const	↓	↓↓	↓	↓	

Table 3 cont.

Operational factor	Designation	Change of TK conditions	Change of TRD parameters							Note
			T_3^*	$R_{уд}$	$G_в$	R	$m_т$	$C_{уд}$	$G_т$	
Hydraulic and gas-dynamic losses										
Losses in the intake duct	$\sigma^*_{вх}\downarrow$	Similarity of TK is preserved	const	↓	↓	↓↓	const	↑	↓	
Losses in compressor	$\eta^*_к\downarrow$	Compressor characteristic is changed	const	↓	↓	↓↓	const	↑	↓	$L_к = const$
Losses in combustion chamber	$\sigma^*_{к.с}\downarrow$		↑	↑	const	↑	↑↑	↑	↑↑	Surge margin is lowered
Losses in turbine	$\eta^*_т\downarrow$	—"—	↑	↑	—"—	↑	↑↑	↑	↑↑	—"—
Losses in the diffuser past the turbine	$\sigma^*_д\downarrow$	—"—	↑	↑—	—"—	↑	↑↑	↑	↑↑	—"—
Losses in afterburner	$\sigma^*_ф\downarrow$	—"—	↑	↑	—"—	↑	↑↑	↓	↑↑	—"—
Losses in jet nozzle	$\varphi_{рс}\downarrow$	—"—	↑	↑	—"—	↑	↑↑	↑	↑↑	—"—
Losses of combustion inefficiency										
Losses in combustion chamber	$\xi_{к.с}$	Similarity of TK is preserved	const	const	const	const	↑	↑	↑	
Losses in afterburner	$\xi_ф$	—"—	—"—	—"—	—"—	—"—	↑	↑	↑	

[Note. TK = turbocompressor]

Increase of losses at the TRD inlet with a supercritical pressure drop in the jet nozzle does not affect the turbocompressor operating conditions, however, it causes a drop of air flow rate, specific and total thrusts, and impairment of the economy of engine operation. With subcritical pressure drop in the jet nozzle the lowering of $\sigma^{*}_{вх}$ decreases the pressure drop on the turbine. Due to this the gas temperature before the turbine is increased, and surge is possible.

Deterioration of compressor efficiency when n = const and $L_{к}$ = const leads to lowering of thrust and increase of specific fuel consumption when $\pi_{р.с}>\pi_{кр}$. When $\pi_{р.с}<\pi_{кр}$ temperature T^{*}_{3} rises and TRD thrust is increased. If the drop of $\eta^{*}_{к}$ causes a proportional rise of $L_{к}$, then as a result there is observed increase of T^{*}_{3}, $R_{уд}$, R and $C_{уд}$.

Impairment of carburetion in the conbustion chamber causes lowering of the combustion efficiency and as a result an increase of specific and hourly fuel consumption. Engine operating conditions are not changed in this case; TRD thrust remains constant.

§2. Thrust Reversing of Turbojet Engines

Continuous increase of maximum flight speeds of passenger and transport aircraft leads to an increase of landing speeds to some degree or another. For the purpose of increasing landing safety this circumstance requires the realization of constructive measures for deceleration of aircraft motion, making it possibe to decrease its approach distance to the airfield, holding before landing and the after-landing on the runway run.

One such measure is the application of so-called thrust reversers.

The device or mechanism, with whose help the engine develops negative thrust, i.e., thrust directed opposite the direction of aircraft motion, is called a thrust reverser.

Besides the creation of negative thrust during deceleration of the aircraft, the reverser provides: making the landing approach without lowering the rpm, which makes it possible when necessary to rapidly restore positive thrust for a go-around; emergency decrease of the speed in flight; increase of maneuverability of the aircraft when taxiing on the ground, and also in flight.

The principle of action of the thrust reverser is based on deflection of the gas jet flowing from the engine.

There are a great many different designs and constructions of thrust reverser, but all of them can be included in two "classic" types:

1. When the flow of gases is deflected with the help of rotary shutters, installed at the jet nozzle exit ("scoop" type reverser).

2. When the flow of gases is deflected by shutters and aerodynamic grids, installed between the turbine and jet nozzle (Fig. 7.4).

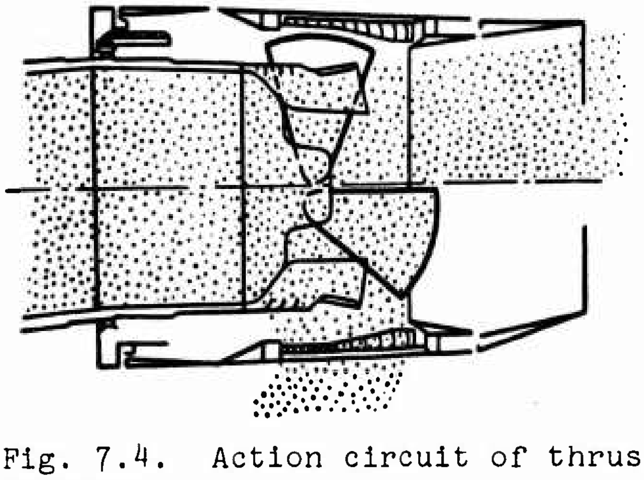


Fig. 7.4. Action circuit of thrust reverser.

The second type of thrust reversers received the greatest application, since they made it possible to obtain a high value of negative thrust.

Thrust reversers during the landing run permit creation of negative thrust equal to 40-50% of maximum stand thrust. This reduces the distance of landing run of the aircraft by 40-60%.

The weight of series-produced thrust reversers comprises around 10% of the engine weight.

Thrust reversers should provide the necessary high-speed operation of change of thrust direction in 1-1.5 s.

§3. Service Life and Reliability of Gas-Turbine Aircraft Engines

Operational characteristics of gas-turbine aircraft engines are determined not only by economical parameters, regularities of change of thrust and fuel consumption depending on the flight speed and altitude, heat release rare of components, characterized by the gas temperature in the "hot" part of the engine, but also by the service life and reliability of the engine.

By lifetime, or service life, we mean the time (in hours) of operation (nonfailure operating time) of the engine without deflects with a definite relationship between the basic operating conditions of the engine: takeoff, nominal and cruise. Service life – this is the nonfailure operating time of an engine between two overhauls.

The engine service life is estabilished by plants (firms)-suppliers, and also by the utilizing organizations (airlines operating the aircraft). The latter change the service life, increasing or lowering it, depending on the concrete operating conditions on a given airway. Thus, for example, if the airway passes through tropical countries with severe operating conditions, then because of the great fall of thrust at high ambient air temperatures the portion of nonfailure operating time at takeoff conditions (i.e., more stressed) increases with respect to the corresponding nonfailure operating time of takeoff conditions on other airways with easier operating conditions.

Consequently, on tropical airways the engine service life should be reduced. The greater the nonstop flying range of an aircraft (i.e., the less the amount of takeoffs), the greater the established engine service life.

The service life of a gas-turbine engine is determined by fatigue phenomena in the construction (in turbine and compressor blades, etc.), appearing under the action of alternating and vibration loads, and also wear of subassemblies.

During the last few years the service lives of gas-turbine aircraft engines guaranteed by aviation firms sharply increased. For example, the service life of engines of the English firm Rolls-Royce comprises: "Avon" - 4000 h, "Dart" - 5000 h, "Tyne" - 3200 h, "Conway" - 7200 h. The Rolls-Royce firm is a distinctive "holder" of world records for the service life of gas-turbine aircraft engines.

On Fig. 7.5 there are given curves of increase of the service life of the above-mentioned four engines of the Rolls-Royce firms versus years. From the figure it follows that setting of service life at 2000 h for contemporary gas-turbine aircraft engines occurs in 2-3 years.

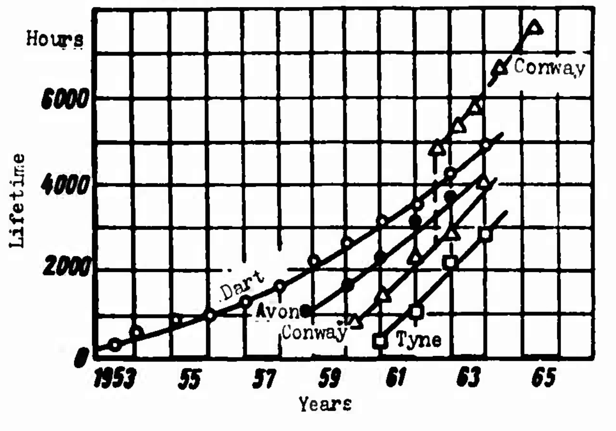


Fig. 7.5. Increase of the service life gas-turbine engines of the Rolls-Royce.

Increase of the service life of aircraft engines is an important factor, making it possible to sharply lower amortization expenditures of aircraft engines and as a result to increase the economy of air transportations.

An important operational characteristic of aircraft engines is their reliability. Reliability of gas-turbine aircraft engines is determined by the probability of their failure in flight operation, for example in flight. It is characterized by the number of hours spent for one ahead-of-schedule engine removal from an aircraft in operation (or for one in-flight failure) or by the reciprocal – the amount of ahead-of-schedule removaed engines (having failed in engine flight), occurring per 1000 h of their non-failure operating time.

Of the series engines well proven in operation the least number of failures is recorded for the ducted-fan engine "Conway" RC_0-12 (0.04 failures in flight, which are apportioned to 1000 h of operation).

CHAPTER VIII

NOISE LEVEL CHARACTERISTICS OF GAS-TURBINE AIRCRAFT ENGINES

Decrease of noise of contemporary gas-turbine aircraft engines is one of the serious problems of civil aviation.

The mass development of air transportation and increase of the power of propulsion systems of contemporary aircraft lead to the fact that the number of persons suffering from noise is sharply increased. Not only are the crew and passengers of the aircraft subjected to it on the ground and in air, but also maintenance personnel in airport areas, and the most important thing — the population of nearby areas.

Noise hampers the normal labor activity of a person, causes premature fatigue, lowers labor productivity, disturbs the normal rest of a person, and can lead to various nervous diseases.

The problem of the struggle with aviation noise has become so much acute that the governments of different countries are forced to introduce special severe limitations of aircraft operation with regard to the time and direction of flights, the permissible level of noise, they are forced to fine aviation companies which exceed the permissible norm for noise.

Table 4 shows the levels of sound intensity of different noises (in decibels).

Table 4. Levels of sound intensity of different noises.

Character and sources of noises	Level of sound intensity of noise dB
Audibility threshold............................	0-10
Rustling of leaves, noise of a gentle breeze....	10-20
An average whisper at a distance of 1 m.........	20-30
A quiet inhabited room..........................	30
Light radio music in an apartment, an inhabited location..................................	40
Restaurant with average business, establishment.	50
An average busy street, a noisy establishment or store....................................	60
Volume range of speech..........................	45-70
Music through a loudspeaker.....................	70-80
Truck...	80
Loud automotive horn at distance of 5-7 m.......	100
An express train traveling at high speed........	110
Reaction engines with overall thrust 4500 kgf at distance of 9 m in the noisiest direction:	
turbojet engine...........................	140
turbojet engine with afterburner............	150
solid-propellant rocket engine..............	155
supersonic prop............................	136
Threshold of painful sensation..................	140
Mechanical damages..............................	160

§1. Sources of Gas-Turbine Engine Noise

A gas-turbine engine has a number of sources of noise. The basic one is the gas stream flowing from the jet nozzle. Being mixed with the environment, it creates intense turbulent fluctuations and at supercritical outflow – a system of shocks,[1] which are powerful generators of noise.

Under aircraft takeoff and climb conditions the pressure drops in the jet nozzle, as a rule, are insufficient for the appearance of

[1]With incomplete expansion of gas, for example in the convergent nozzle.

strong shock waves. Therefore, at takeoff the noise level of the emanating jet is determined basically by turbulent fluctuations. At cruise flight performance intense noise can be generated by the shock waves and turbulent fluctuations. Appearing vortex formation, in which the kinetic energy of the jet is dispersed, converting into heat, generate pressure fluctuations and the latter are sources of noise.

Turbulent mixing of the jet with environment encompasses a region, the axial length of which is equal to 15-25 diameters of the nozzle. In this region there is generated practically all the noise of the jet flowing from the engine (Fig. 8.1).

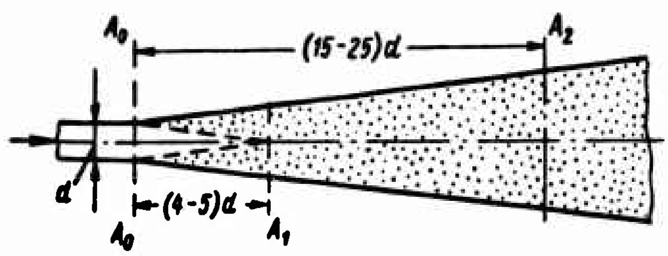


Fig. 8.1 Diagram of a free turbulent stream.

Another powerful source of noise is the revolving compressor, and also acoustic interaction of the streams circumflowing the rotor and stator blades. Around every blade a field of pressures appears. If the peripheral velocity of blades is great, then the field of pressures will pass through the compressor intake into the free space in the form of a wave with increasing intensity.

The noise level of the compressor (fan) is usually lower than the noise level of the stream flowing from the engine, however, it is characterized by high-frequency pressure fluctuations, "whistle," which has a most unpleasant physiological effect on a person.

A source of noise in a turboprop engine is also the rotating propeller. In this case there appear so-called vortex noise, caused

by vortices periodically separating from the prop blade and rotation noise, generated by fluctuations of pressure and speed near the plane thrown by the prop. These fluctuations are connected with displacement of air by the blades and the formation of a pressure differential on both sides of the blade. The noise level of the propeller the greater, the higher the M number at the blade tip, the smaller the number of prop blades and the greater the input power to the prop.

During normal operation of a gas-turbine engine the noise appears furhtermore, in the turbine, and also in the combustion chamber because of irregular turbulent burning. However, it is usually concealed by noise of the outflowing stream.

During finishing and testing of afterburners, and also during their operation a special type of pulsational burning appears in some cases — so-called "resonance" burning. The latter is accompanied by sharp "squealing" sound, resembling a pipe organ.

§2. Evaluation of the Noise Level of the Outflowing Stream

In accordance with the experimentally proven Lighthill theory the acoustic power of noise of an outflowing subsonic free turbulent stream is determined by formula

$$N = k\frac{\rho_5^2 d_5^2 c_5^8}{\rho_H a_H}\ [\text{kgf-m/s}], \qquad (8.1)$$

where d_5 — diameter of jet nozzle exit section; ρ_5, c_5 — density and velocity of outflowing gas; ρ_H, a_H — ambient density and the speed of sound in it; k — proportionality factor, determined experimentally.

The power of noise is determined basically by the exit velocity of gas and is proportional to its value in the eight power.

In technology as the basic characteristic of noise we use the parameter of its intensity level

$$L = 10 \lg \frac{J}{J_0} \text{ [dB]}, \tag{8.2}$$

where J — the sound intensity of noise on the surface of hemisphere of radius r, in the center of which is the source of noise

$$J = \frac{N}{2\pi r^2};$$

J_0 — sound intensity at the audibility threshold.

Depending on the gas exit velocity the level of noise is depicted graphically by logarithmic curve

$$L = 80 \lg c_5 + 10 \lg A, \tag{8.3}$$

where

$$A = f(d_5, \rho_5, r \ldots).$$

Dependence of the noise level of certain contemporary engines with thrust R ≈ 5000 kgf on the gas exit velocity from the nozzle is shown in Fig. 8.2.

Possible differences in the density and temperature of the outflowing stream at a prescribed exit velocity given noise fluctuations on the order of 2-3 dB. From Fig. 8.2 it follows that if at a distance of 250 m from an aircraft the subsonic turbojet engine has noise level of 118 dB with exit velocity of the stream 600 m/s, then at exit velocity 360 m/s a ducted-fan engine has noise level of 103 dB, i.e., 15 dB lower. Boosted turbojet engines at exit velocity 720 m/s have noise level of 124-125 dB.

The reaction of a person to noise depends not only on its physical level, determined by sound pressure in decibels, but on an entire series of factors, including frequency-response curve (spectrum) of noise, its duration, monotony or shock of action, etc.

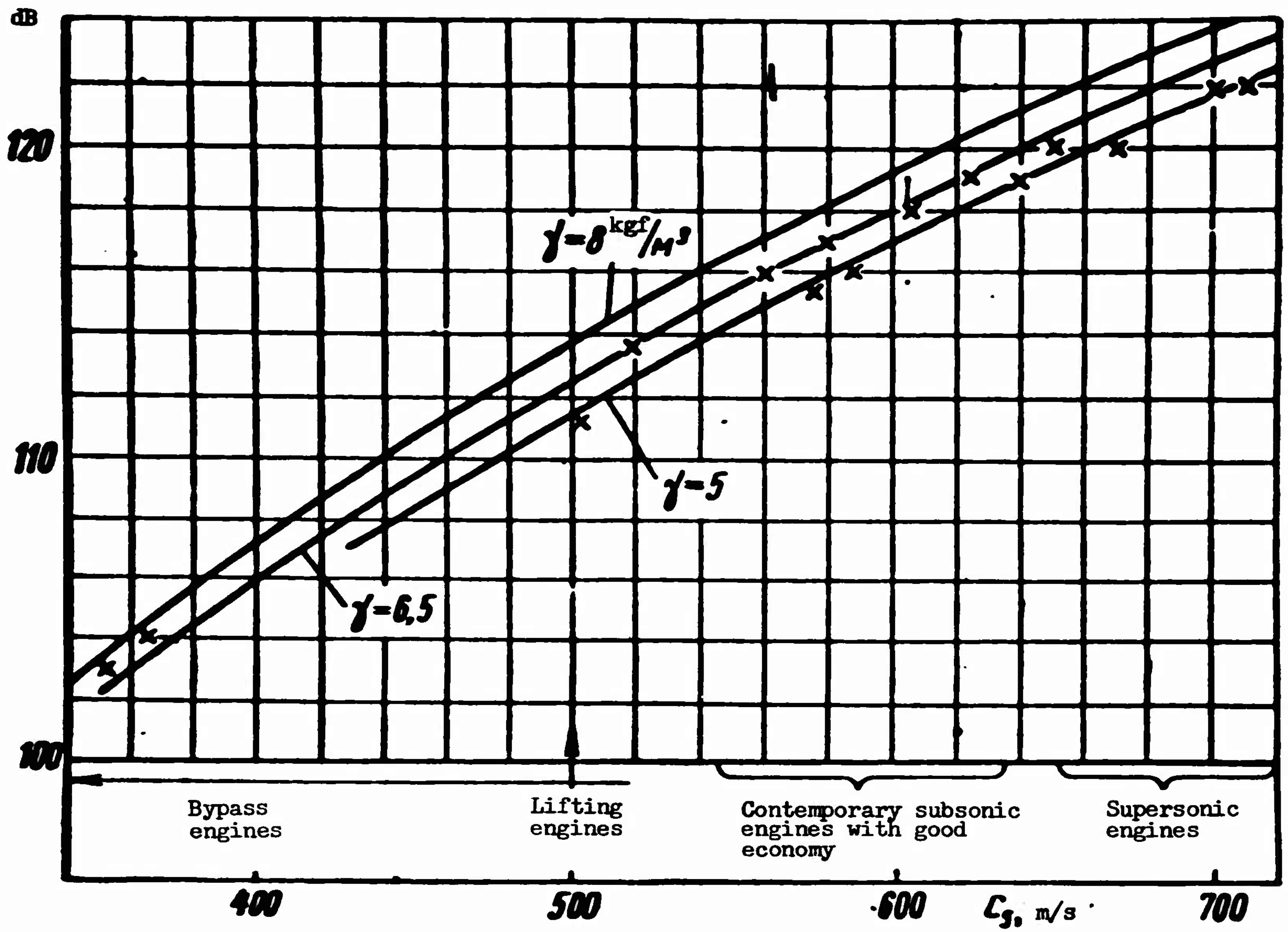


Fig. 8.2. Effect of the exit velocity from the jet nozzle of a turbojet engine (ducted-fan engine) on the noise level (at a distance of 250 m from the source of noise).

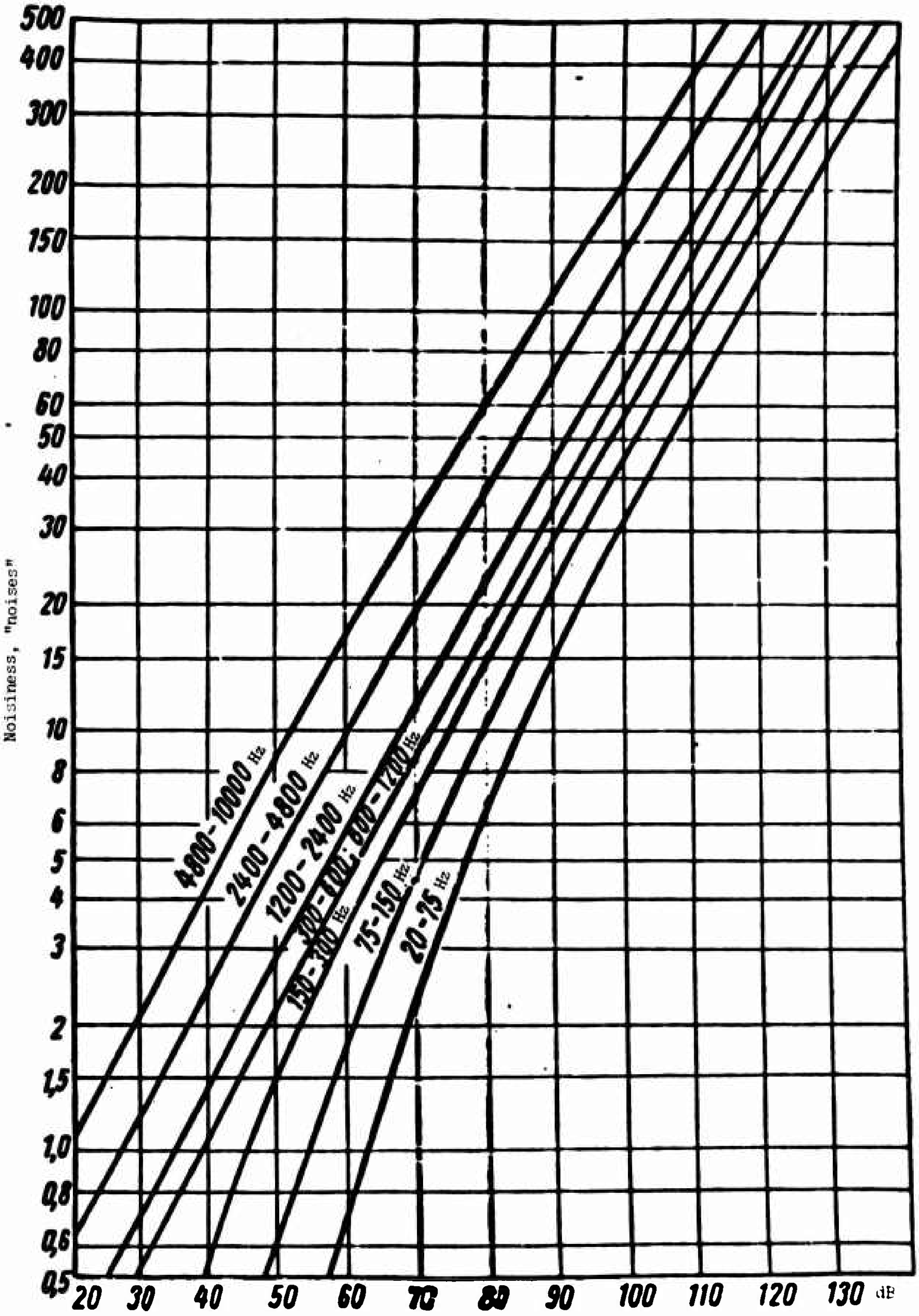


Fig. 8.3. Dependence of noisiness in "noise" on the level of sonic pressure in the octave.

As a result of performing special experiments with the participation of many people at present there is introduced a new method of evaluating noisiness with the aid of "noise" – unit of perceived noise [Translators Note: the word нойз (noise) was transliterated from English to Russian], designated PN[1] decibel. One "noise" is equal to the noisiness of the octave band 600-1200 Hz[2] of arbitrary noise at the level of sonic pressure in 40 dB.

Change of the frequency of noise leads to change of the level of perceived noise. On Fig. 8.3 there is represented the dependence of of noisiness in "noises" on the level of sonic pressure in the octave. The higher the frequency of noise, the greater is the perceived level of noise in "noises."

Figure 8.4 contains a spectrum of noise, obtained on the ground from a DC-8 aircraft flying at an altitude of 800 m. When the

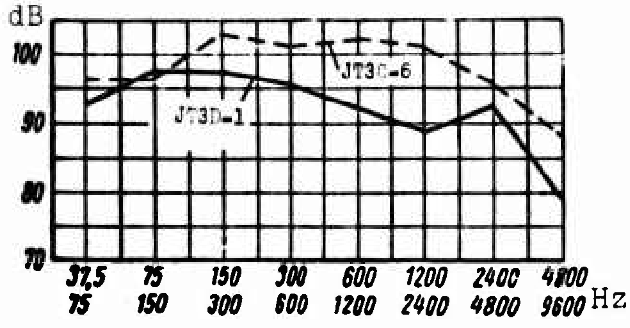


Fig. 8.4. Spectrum of noise of DC-8 aircraft with single-spool and bypass engines.

aircraft is equipped with single-spool JT3C-6 turbojet engines, its overall noise[3] level is 110 dB, and perceived noise – 120 PN dB, if

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[1]PN – perceived noise.

[2]Band of the greatest sensitivity of the ear.

[3]Determined by integral power of noise for the entire frequency spectrum.

the aircraft is equipped with JT3D-1 ducted-fan engines, – the noise levels are 104 dB and 114.5 PN dB respectively.

Treatment of experimental data shows that the level of perceived noise of a stream, as a rule, exceeds the level of sonic pressure by 8-10 dB.

§3. Methods of Lowering the Noise Level

Various methods of lowering the noise level exist. They include:

application of special noise suppressors of the jet stream, including ejector type; rational location of the engine on the aircraft;

application of ducted-fan engines – engines with a lowered outflowing stream velocity; application of acoustic grids (barriers) in the air intakes of engines or aircraft; rational selection of the aircraft takeoff profile.

Application of Jet Stream Noise Suppressors

The principle of the muffler device is based on the splitting of one powerful stream, exiting the engine, into many small streams. Acoustic interference (interaction) between mixing zones of the broken-up streams leads to lowering of their total noise level.

Noise suppressors (noise-suppressing nozzles) have different geometric configuration and construction – tubular, lobe, lobe with a centerbody, corrugated nozzle, ejector nozzle, etc., (Fig. 8.5). Such noise suppressors, as a rule, are used on single-spool turbojet engines (for example "Avon," Pratt and Whitney JT3C-6 and others). With their help the noise level of the exhaust stream of a turbojet engine is lowered by approximately 3-5 dB. Only the application of more complicated corrugated nozzles with an ejector permits decreasing the noise level by 12-15 dB.

Fig. 8.5. Mechanical noise suppressor

Disadvantages of the noised-suppressing nozzles is increase of weight of the propulsion system (by 2-3%), lowering of engine thrust (by 2-4%), increase of external drag of the engine nacelle (due to increase of wake drag), impairment of engine economy.

By rationally selecting the configuration of the noise-suppressing nozzle, it is possible to achieve displacement of acoustic energy to the zone of high-frequency oscillations. Such noise, as it is known, attenuates rapidly at great distances from the nozzle.

On the whole one should note that a satisfactory solution for lowering the noise level by means of noise suppressors has as yet not been found.

Rational Location of Engines on the Aircraft

Location of the engines in one plane gives the possibility of considerably lowering the noise level developed by them. The effect of lowering the noise level in the case is explained by acoustic shielding and the interaction of mixing zones of the streams. Replacement of one powerful engine by a series of engines (of equal total thrust) can lower the overall level of noise by 5-15 dB.

Effect of the Bypass Ratio of Ducted-Fan Engines on the Level of Noise.

Increase of the bypass ratio at constant working process parameters of the initial turbojet engine leads to a continuous lowering of the exit velocity from the ducts, and consequently, to lowering of the noise level.

Figure 8.6 shows the effect of the bypass ratio on the gas exit velocity from the ducted-fan engine with optimum distribution of energy. As initial turbojet engine there is taken an engine with exit velocity c_5 = 600 m/s. With increase of y the average gas exit velocity is continuously decreased:

when $y=0$ $c_5=600$ m/s;
$y=1$ $c_5=400$ m/s;
$y=2$ $c_5=320$ m/s;
$y=6$ $c_5=200$ m/s.

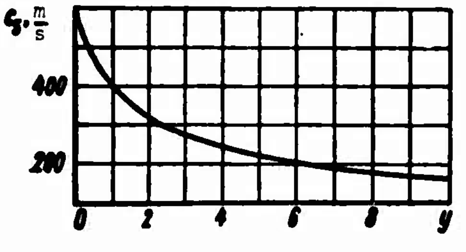

Fig. 8.6. Effect of the bypass ratio on the average gas exit velocity from a ducted-fan engine

On Fig. 8.7 there is shown the effect of the bypass ratio and the gas temperature before the turbine on the relative noise level of a ducted-fan engine. From the figure it is clear that increase of y from 0 to 1.5 lowers the noise level by 20 dB. At the same time, lowering of T_3^* from 1350 to 1200°K (i.e., by 150°) reduces the noise level by only 5 dB.

For the purpose of lowering the level of engine noise and the operational flow rates it is expedient with great flying range to

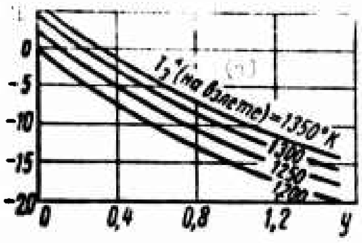

Fig. 8.7. Effect of the bypass ratio and gas temperature before the turbine on the relative noise level of a ducted-fan engine.
KEY: (a) At takeoff.

apply high-temperature ducted-fan engines with high bypass ratio without any additional devices for lowering the level of noise. As compared to a turbojet engine this provides a considerable economic effect and practically solves the problem of noise abatement.

Mixing the flows in an unboosted ducted-fan engine additionally lowers the noise level (by 3-4 dB) due to equalizing the velocity profile in the exhaust stream (Fig. 8.8).

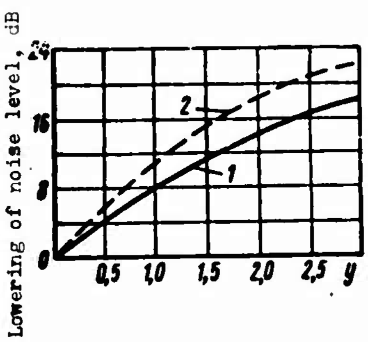

Fig. 8.8. Effect of mixing of flows in a ducted-fan engine on the noise level: 1 – without mixing of flows; 2 – with mixing of flows.

Effect of the Bypass Ratio on the Noise Spectrum.

Increase of the bypass ratio not only leads to lowering of the noise level. It, furthermore, changes the frequency-response curve of the noise spectrum, shifting the maximum of its level to the region of low-frequency oscillations.

In this case perception of the same level of noise by the human ear becomes more favorable, since it is sensed as a weaker sound (noise).

On Fig. 8.9 there are depicted curves of noise spectra, created by turbojet engine CJ-805-3 without noise suppression and with a

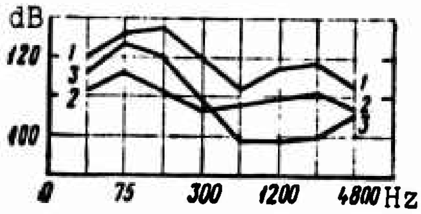

Fig. 8.9. Curves of noise spectra of turbojet and ducted-fan engines: 1 – turbojet engine CJ-805-3 without noise suppression 2 – turbojet engine CJ-805-3 with noise suppressor; 3 – ducted-fan engine CJ-805-23.

noise suppressor, and also a ducted-fan engine CJ-805-23 with assigned location of the fan. From the figure it is clear that in the region of high-frequency oscillations the CJ-805-23 engine has the lowest noise level (10 dB lower than for turbojet engine CJ-805-3 without noise suppression). This advantage turns out to be very considerable if we consider that the application of a turbofan attachment when y = 1.5 increases the takeoff thrust of an engine by 35-40%.

Comparison of the Noise Levels of a Lifting Turbofan Assembly and Compressor.

Application of wing lifting turbofan assemblies TVA with very low gas (air) exhaust velocities from the spools at large values of y gives the possibility of very significantly lowering the noise level. This is shown visually on Fig. 8.10, from which it follows that when c_5 = 180-200 m/s the noise level of the emanating gas stream from the TVA does not exceed 105-107 dB.

Further lowering of the exit velocity from the TVA, caused by the tendency to decrease the noise level still more, is no longer justified due to limitations imposed by the engine compressor and TVA fan. Really, considerations of decrease of engine dimensions and weight compel the designers to apply compressor and fan stages with high axial velocity at the intake, reaching 200 m/s and more. This increases the level of noise at the compressor (fan) inlet. Thus, the level of noise 100-105 dB, determined by "whistle" of the compressor (fan), is apparently now lower than the noise limit of a contemporary ducted-fan engine. For lowering the noise level of the fan a stator should not be installed in front of it.

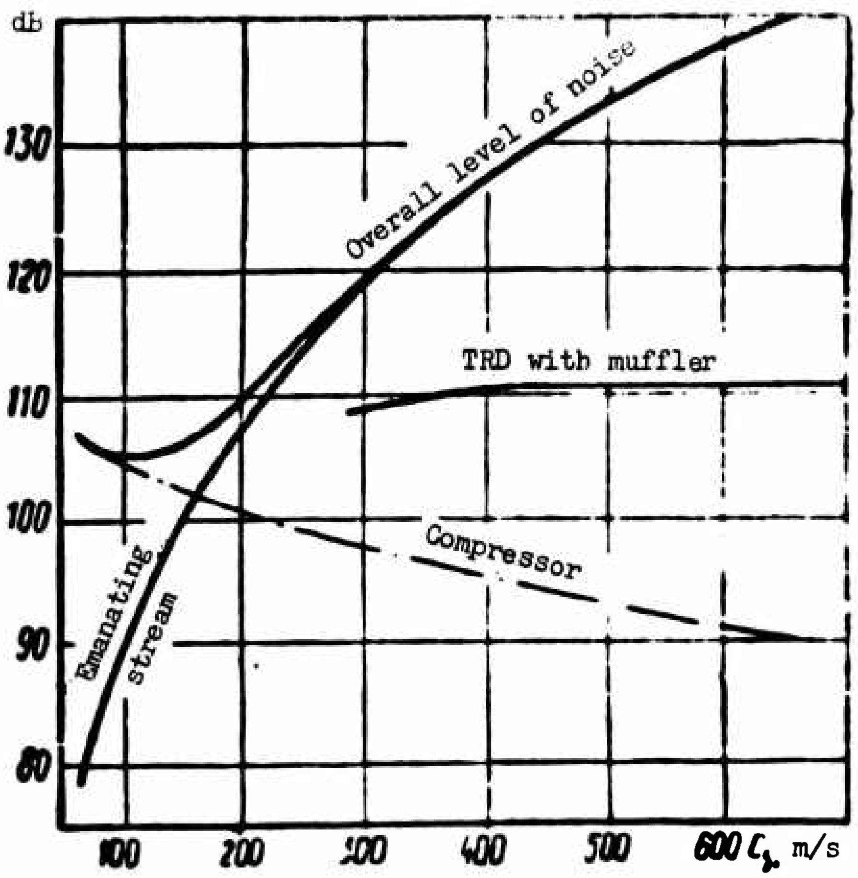

Fig. 8.10. Comparative curves of noise level for the emanating gas stream and the compressor of a ducted-fan engine.

Application of Acoustic Grids in Air Intakes of the Engines or Aircraft

It is possible to prevent the "radiation" of fan noise into the environment if an acoustic grid is installed in the air intake of the engine or aircraft. The grid either damps the acoustic energy of the revolving compressor or does not "pass" sound vibrations upwards along the inlet flow. According to foreign reports such devices are applied on the ducted-fan engine JT-8D-1 of the Pratt and Whitney firm.

Rational Selection of the Aircraft Takeoff Profile

The maximum level of perceivable noise under the aircraft in an inhabited region at a distance of 5-6 km from the beginning of an

airfield runway should not exceed a certain permissible limit. As such a limit in the United States there is established a day norm of 112 dB and night 102 dB.

Rational selection of the aircraft takeoff profile can be assured by observance of these norms on the border of the airfield even with insufficiency of noise-suppressing means.

On Fig. 8.11 there is shown the takeoff path of a transport aircraft, equipped with gas turbine engines, with speed of acceleration (after takeoff) up to 480 km/h (curve a).

During a mildly sloping takeoff with engines operating at maximum power, on the border of the airfield (takeoff distance L = 5.6 km, H = 100 m) the perceivable level of noise is equal to 121 dB. It is lowered to 107 dB in proportion to further climb (L = 8.8 km, H = 360 m).

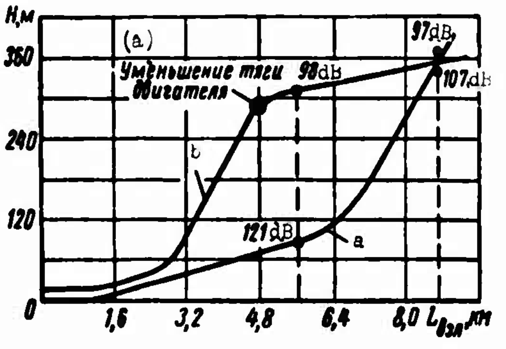

Fig. 8.11. Effect of the aircraft takeoff path on the level of noise at the control point.
KEY: (a) Reduction of engine thrust.

It is more rational to accelerate the aircraft to a lower speed, for example 290 km/h, and to climb to 300 m at maximum engine thrust (curve b). After achievement of this altitude the pilot reduces the engine thrust (up to 50%) and decreases the climb angle, while maintaining a flight speed of 290 km/h.

In this case at flight through a control point (L = 5.6 km, H = 300 m) the noise level is equal to only 98 dB, i.e., 23 dB less than with usual mildly sloping takeoff path.

It is necessary to note that of the two methods of throttling engine thrust (decrease of revolutions at f_5 = const and opening the jet nozzle when n = const), the second is more preferable; at equal lowering of thrust it is characterized by a lower noise level of the emanating gas stream (Fig. 8.12).

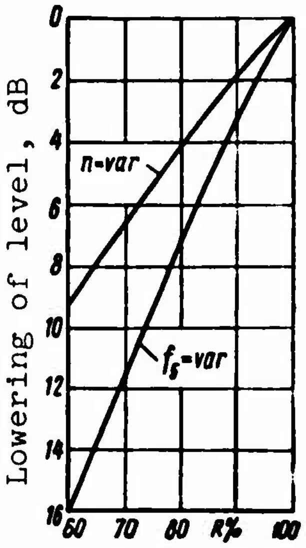

Fig. 8.12. Comparison of two method methods of throttling turbojet engine thrust with respect to the noise level.

CHAPTER IX

STARTING AND ACCELERATION CHARACTERISTICS OF GAS-TURBINE ENGINES

§1. Gas-Turbine Engine Starting

Starting a gas-turbine engine is possible only with the aid of an outside source of energy that is powerful and effective for a relatively prolonged interval of time. In this respect the starting of a gas-turbine engine essentially differs from starting a reciprocating engine, for which motoring of the crankshaft manually or mechanically 2-3 revolutions is sufficient. For a gas-turbine engine, starting only from a specific, sufficiently high number of revolutions (let us call them idle revolutions), at which there is created raised outlet pressure from the compressor, stable work of the combustion chamber turns out to be possible, and the turbine can develop excess power for rotation of the compressor.

At lowered conditions, when gas-turbine engine revolutions are lower than idle revolutions, autonomous operation of the gas-turbine engine is impossible.

Devices which produce the power necessary for starting and spinup the gas-turbine engine rotor to idle revolutions are called starters.

For starting the gas-turbine engines there are applied piston, electrical, gas-turbine, air and cartridge starters.

The piston starter is a small two-cycle or four-cycle internal-combustion engine, actuated in turn by an electric starter. Such a starter ("Ridel'" type) was applied on the first turbojet engines YuMO-004 and BMV-003 with 700-900 kgf thrust. It developed up to 12-15 hp.

A deficience of the piston starter is its excessively high specific gravity and deminsons. Furthermore, it is difficult to assure its reliable operation in winter.

Electric starter – more effective means of starting. It is successfully utilized on turbojet engines of medium thrust (up to 2000-3000 kgf). The electric starter is a generator with parallel or mixed excitation. It is supplied from airborne or airfield storage batteries with voltage up to 48 V. It is simple in operation, and permits fully automating gas-turbine engine starting. However, a big disadvantage of it is the large weight of storage batteries, reaching 120-180% of the generator weight, and also a sharp drop of their capacity with lowering of the ambient temperature. Specific gravity of a contemporary electric starter is 5-8 kgf/hp.

Gas-turbine starter received very wide application on contemporary gas-turbine aircraft engines with great thrust. This is a small gas-turbine engine, transmitting power from the turbine to the shaft of the rotor of the gas-turbine engine to be started. The gas-turbine starter is started by an electric starter.

The gas-turbine starter can develop very considerable power (up to 300-400 hp) with comparatively low weight and small dimensions. Specific gravity of the starter is 1.3-1.8 kgf/hp.

The gas-turbine starter with a free turbine received wide propagation.

A deficience of the gas-turbine starter is structural complexity and the duration of turbojet engine starting (including the time of starting the gas-turbine starter and its electric starter).

Air starter has a free turbine, rotated by compressed air. The turbine is equipped with a mechanism for coupling with the turbo-compressor shaft of the gas-turbine engine. Compressed air is removed from a special air-generator, being a turbine engine, for whose excess turbine power is expended for compression of an additional quantity of air in a special centrifugal compressor.

The system of air starting, as compared to others, is characterized by the least weight and the greatest reliability. It received wide propagation, in particular, on American military and transport gas-turbine engines.

Dynamics of Gas-Turbine Engine Starting

Fundamental Equation of Gas-Turbine Engine Starting.

Starting of a gas-turbine engine is a transient process; it can be described by the following equation of dynamics:

$$J\varepsilon = \Delta M = M_{\text{ст}} + M_{\text{т}} - M_{\text{к}} - M_{\text{тр}}. \tag{9.1}$$

Here $M_{\text{ст}}$ – torque of the starter; $M_{\text{т}}$, $M_{\text{к}}$ – torques of the turbine and compressor respectively; $M_{\text{тр}}$ – moment expended for overcoming mechanical losses (in bearings) and for driving the assemblies; ΔM – excess moment, required for acceleration of the engine rotor; J – polar moment of inertia of the rotor; $\varepsilon = \frac{d\omega}{dt}$ – angular acceleration of the rotor.

Having replaced torques by powers with the aid known relationships

$$\omega = \frac{\pi n}{30} \text{ and } N = \frac{M\omega}{75},$$

we obtain the fundamental equation of starting:

$$Jn\frac{dn}{dt} = \frac{75 \cdot 900}{\pi^2}(N_{ст} + N_т - N_к - N_{тр}). \quad (9.2)$$

Starting Stages.

The process of starting a gas-turbine engine can be subdivided into three stages (Fig. 9.1).

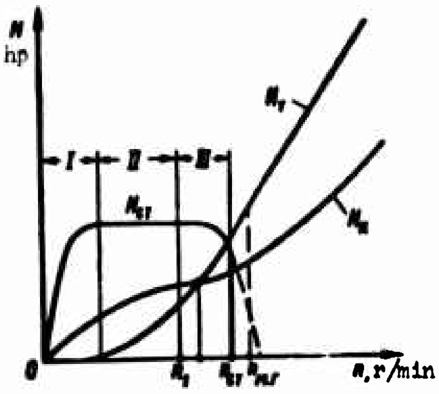

Fig. 9.1. Engine starting stages.

First stage – GTD combustion chamber does not work and the turbine does not develop power; it is described by equation

$$Jn\frac{dn}{dt} = 6840(N_{ст} - N_к - N_{тр}), \quad (9.3)$$

in the case $N_т = 0$.

Second stage – GTD combustion chamber and turbine operate; equation of starting has the form

$$Jn\frac{dn}{dt} = 6840(N_{ст} + N_т - N_к - N_{тр}), \quad (9.4)$$

in this case $N_т < N_к + N_{тр}$.

Third stage - starter is disconnected, and GTD rotor reaches idle revolutions. This stage is described by the following equation:

$$Jn\frac{dn}{dt}=6840(N_{\text{т}}-N_{\text{к}}-N_{\text{тр}}), \tag{9.5}$$

in this case $N_{\text{ст}}=0$, $N_{\text{т}}>N_{\text{к}}+N_{\text{тр}}$.

Cutoff of the starter usually occurs at revolutions considerably higher than those (n') at which turbine and compressor powers are equilized $(N_{\text{т}}=N_{\text{к}}+N_{\text{тр}})$:

$$n_{\text{ст}}=(1{,}5\div 2)\,n'.$$

Starting the Gas-Turbine Engine Flight

A peculiarity of GTD starting in flight involves the fact that here it is not necessary to spinup the engine rotor with the aid of the starter. The relative wind rapidly rotates the rotor (windmilling conditions), which makes ignition of fuel in the combustion chambers possible.

It is necessary to bear in mind that under windmilling the compressor develops almost no excess pressure. The necessary increase of pressure for effective combustion is assured by ram pressure. Therefore operating conditions of the combustion chamber for a windmilling engine are less favorable than in the case of normal ground starting.

When starting in flight with climb to altitude flameout is possible. To ensure reliable high-altitude starting under windmilling conditions it is necessary to provide for the formation of a powerful flame kernel in the combustion chamber.

The Effect of External Atmospheric Conditions on Gas-Turbine Engine Starting

With lowering of ambient temperature the friction power increases.

Due to this the excess power of the starter is lowered, starting is "delayed," idle revolutions are lowered, and conditions of acceleration of the rotor to operating conditions worsen. Furthermore, the power of the compressor grows in accordance with increase of the given number of revolutions.

§2. Transient Conditions of Gas-Turbine Engines

After the process of starting is completed and the turbojet engine is brought to idle, further acceleration of the turbojet engine to maximum revolutions is carried out only by increasing the fuel feed to the combustion chamber. This is produced by smooth movement of the engine control lever, interlinked with the automatic fuel feed control, to the "stop." During fuel feed to the combustion chamber the gas temperature before the turbine increases, the turbine power turns out to be greater than compressor power ($N_т > N_к$), in connection with which the turbocompressor revolutions continuously increase.

With reduction of fuel feed by shift of RUD to the idle stop, conversely, the gas temperature before the turbine drops, the turbine power turns out to be lower than compressor power ($N_т < N_к$) and compressor revolutions are lowered accordingly from maximum to minimum (at idling).

Acceleration and deceleration processes of the turbocompressor are transient conditions of engine operation. They are described by equation of motion

$$Jn\frac{dn}{d\tau}=\frac{900\cdot 75}{\pi^2}(N_т-N_к). \qquad (9.6)$$

On Fig. 9.2 there is given line AA'mB of the combined operating conditions of compressor and turbine of the dynamic transient conditions of acceleration of the TRD turbocompressor. As can be seen, it essentially differs from line ACB of operating conditions of steady conditions, and it is better to say quasi-static process, with which we replace an infinitely large totality of equilibrium states of the

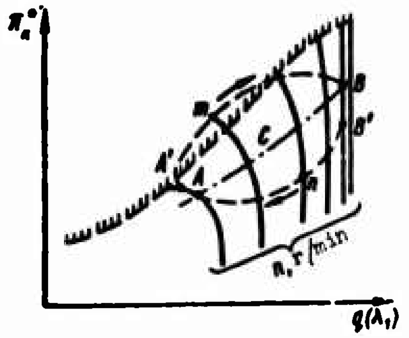

Fig. 9.2. Comparison of dynamic lines of operating conditions of the turbocompressor with increase and decrease of fuel feed.

turbocompressor at various numbers of revolutions. On the figure there is also given the line of the dynamic process of deceleration of the TRD turbocompressor (BB'nA).

Since with increase of fuel feed to the combustion chamber in the first instant the rotor revolutions remain constant, and the gas temperature sharply increases, then the dynamic behavior point of the compressor shifts along its pressure characteristic (n = const) in the direction of the surge line. In proportion to accumulation of excess shaft horsepower of the turbocompressor, the latter accelerates, but with some lag. In this case the dynamic line of operating conditions is deformed, as is shown on Fig. 9.2, gradually approaching the line of static behavior.

With decrease of fuel feed ("jettison") in the first instant the rotor revolutions also remain constant, temperature T_3^* is sharply lowered. In this case the dynamic behavior point of the compressor shifts along line n = const, increasing the surge margin. Then the line of dynamic behavior intersects the line of static behavior in the idle region.

Thus, during acceleration of the rotor the line of operating conditions of a turbojet engine passes in the zone of unstable compress compressor operation and impermissibly high gas temperatures T_3^*. With deceleration of the rotor it passes in the zone of very low values of T_3^*, which can lead to cessation of burning in the combustion chamber.

For preventing these undesirable phenomena in TRD operation and approach of dynamic lines of operating conditions to static we apply

to the combustion chamber in accordance with increase of air pressure past the compressor. In this case the dynamic lines of operating conditions hardly differ from static line ACB.

§3. Acceleration of TRD (DTRD)

By acceleration we mean the ability of the engine during fuel feed to raise from idle to maximum revolutions. Accordingly, the time, required for acceleration of the engine from idling to maximum performance is called the acceleration time. For the majority of single-shaft turbojet engines if is equal to 15-20 s. For new two-shaft ducted-fan engines the acceleration time does not exceed 10-12 s.

For the purpose of determining the factors affecting acceleration time, let us write the equation of dynamics of TRD acceleration in the form:

$$Jn\frac{dn}{dt}=\frac{900\cdot 75}{\pi^2}(N_{\text{т}_{\text{макс}}}-N_{\text{к}})=\frac{900G_{\text{в}}}{\pi^2}(L_{\text{т}_{\text{макс}}}-L_{\text{к}}),$$

whence

$$\Delta t_{\text{пр}}=\frac{\pi^2}{900}\int_{n_{\text{м.г}}}^{n_{\text{макс}}}\frac{Jndn}{(L_{\text{т}_{\text{макс}}}-L_{\text{к}})G_{\text{в}}}. \qquad (9.7)$$

From equation (9.7) it follows that the acceleration time depends on the moment of inertia of the engine rotor, number of revolutions of the rotor, the flow rate of air through the engine and excess power (work) of the turbine. The acceleration time is less, the smaller the moment of inertia of the GTD rotor, the greater the excess power (work) of the turbine (i.e., the higher the gas temperature before the turbine and the pressure drop on it), and the greater the flow rate of air through the engine.

For improvement of TRD acceleration, i.e., lowering of its acceleration time, it is necessary to reduce the moment of inertia of

the masses to spinup (by transition from single-shaft to two-shaft engines, application of light materials in the turbocompressor construction), to increase the maximum allowable gas temperature before the turbine by the use of materials of raised strength for blades, to increase the pressure differential on the turbine by opening the jet nozzle under starting conditions. Use of these means in a number of cases permits lowering the acceleration time to 10-12 s, and for lifting engines – up to 4-5 s.

Figure 9.3 shows the acceleration curve of ducted-fan engine "Spey" during flight near the ground and at altitude.

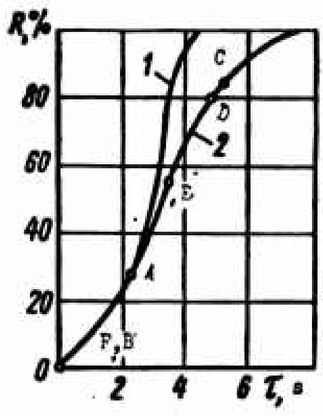

Fig. 9.3. Acceleration curves of ducted-fan engine "Spey" near the ground (1) and at altitude H = 7600 m (2): A – landing approach (n = 10600 r/min); B – touch-down (n = 7500 r/min); C – maximum rate of climb (n = 11900 r/min); D – maximum cruise setting (n = 11770 r/min); E – usual cruise setting (n = 11200 r/min); F – idling flight (n = 9000 r/min).

Effect of Flight Altitude on the Acceleration Time of Turbojet Engines

With climb to altitude the mass flow rate of air through the engine is decreased, in connection with this the excess power of the turbine drops and the acceleration time increases.

At high altitudes (8-11 km and above) the acceleration time is increased 4-5 times as compared to the stand. Therefore, at the indicated altitudes it is not recommended to considerably throttle the engine. The maneuvering properties of turbojet engines at altitude are impaired because of increase of the minimum revolutions of stable engine operation.

Comparison of Acceleration of Reciprocating and Gas-Turbine Engines

Acceleration of a reciprocating engine is considerably better than a gas-turbine. From the practice of operation it is known that a reciprocating engine reaches maximum performance in 2-3 s. This is explained by the fact that for a reciprocating engine in the entire range of possible conditions the excess power, i.e., the difference between available power on its external characteristic and required (propeller) power on throttle characteristic reaches very high values (Fig. 9.4b). At the same time, for a turbojet engine the excess turbine power above the required compressor power is relatively small (Fig. 9.4a) because of the impossiblity of substantially increasing the gas temperature before the turbine.

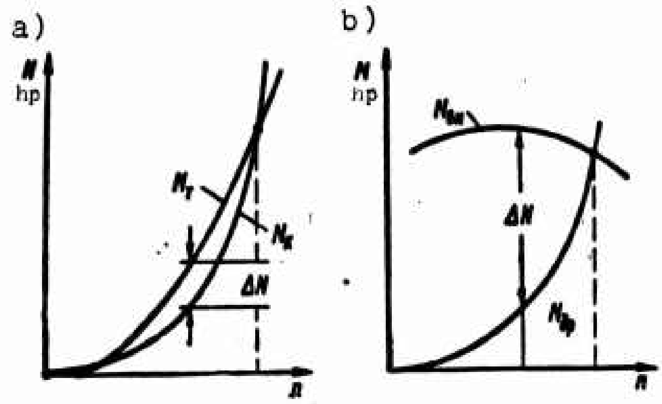

Fig. 9.4. Comparison of acceleration of gas-turbine (a) and reciprocating (b) engines.

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